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AN ASSESSMENT OF THE READINESS OF ABLATIVE MATERIALS FOR PREFLIGHT APPLICATION TO THE SHUTTLE ORBITER

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AN ASSESSMNET OF THE READINESS OF ABLATIVE MATERIALS FOR PREFLIGHT APPLICATION TO THE SHUTTLE ORBITER

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ABSTRACT

The shuttle orbiter relies primarily on a reusable surface insulation (RSI) thermal protection system (TPS). This RSI TPS was selected primarily to reduce operating costs and to minimize turn around time between launches. The RSI is very efficient in its thermal performance, so it provides a light weight TPS. However, the RSI tile system has shown poor mechanical integrity. In view of the apparently random features of the tile integrity problem, the dimension of the required effort to improve the integrity cannot be specified. Therefore, an investigation is needed of other heat shield systems which might be used to replace RSI on the shuttle for one or more flights.

The ablative systems are far more highly developed than other alternatives, and are the only systems that can be considered for near term replacement of RSI.

The purpose of this paper is to review the state-of-the-art of ablative TPS by reviewing the work done as part of the shuttle technology program and to assess the readiness of ablators for use on the shuttle orbiter. Unresolved technical issues with regard to ablative TPS on shuttle are identified and the tasks NASA Langley Research Center (NASA/LaRC) proposes to perform to help resolve the areas of concern are outlined.

The NASA/LaRC initiated a task to examine some of these areas of concern which included short time, highly focused analytical and experimental programs to: (1) identify candidate ablation materials; (2) assess the data base for these materials; (3) evaluate the need and kind of waterproof coating; (4) calculate thermal and other stresses in an ablator tile; (5) identify an acceptable ablator/RSI tile joint filler; and (6) assess the sensitivity of the ablator to sequential heat pulses. Two ablation materials have been identified for use on the shuttle: the Viking heat shield material and the PRIME heat shield material. The PRIME material would be used where recession of the Viking material would be unacceptable. Preliminary calculations showed that replacing an RSI tile with an equal thickness of Viking material would not permit back-surface temperature to exceed 810°R. The supply of this material reinforced with the honeycomb used in earlier programs is essentially nonexistent so that a data base may have to be generated. Data have been compiled for both thermophysical and mechanical properties, however, their statistical basis is not known. A complete validated mechanistic surface recession model for these materials is needed for reliable flight predictions. The weakest link in the data base is the availability of mechanical property data for both materials.

INTRODUCTION

The shuttle orbiter relies primarily on a reusable surface insulation (RSI) thermal protection system (TPS). This RSI TPS was selected primarily to reduce operating costs and to minimize turn around time between launches. The RSI is very efficient in its thermal performance, so it provides a light weight TPS. While these considerations provide sound justification for the selection of an RSI heat shield, in the shuttle TPS as implemented, the RSI tile system has shown poor mechanical integrity. Currently, a major effort is directed toward improving the integrity of the RSI system. However, in view of the apparently random features of the tile integrity problem, the dimension of the required effort cannot be

other heat shield systems which might be used to replace RSI on the shuttle for one or more flights.

The alternative TPS are metallic heat shield systems and ablative materials. Of these, the ablative systems are far more highly developed, and are the only systems that can be considered for near term replacement of RSI. They have been researched and developed to a greater extent than any other kind of thermal protection. They have been tested extensively on the ground as well as in flight projects such as Mercury, Gemini, Apollo, PRIME, Viking and numerous smaller flight programs. In all these applications, the ablative TPS was proven reliable and efficient. Prior to and after the decision to consider only the RSI TPS for the orbiter, ablative TPS design and development studies were conducted at NASA Langley Research Center in the late 1960's through the early 1970's. The results of many of these studies are presented in references 1-29.

The purpose of this paper is to assess the state-of-the-art of ablative TPS by reviewing the work done as part of the shuttle technology program and to assess the readiness of ablators for use on the shuttle orbiter. Unresolved technical issues with regard to ablative TPS on shuttle are identified and the tasks NASA/LaRC proposes to perform to help resolve the areas of concern are outlined.

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SYMBOLS

The units for the physical quantities used herein are given inthe U.S. Customary Units. Appendix A is included for the purpose of conversion to the International System of Units.

A pre-exponential factor for pyrolysis reaction

B pyrolysis activation temperature

C_p specific heat

h_e enthalpy

k thermal conductivity

m_p pyrolysis rate

P_t total pressure

q_c convective heat trasnfer rate

$\bar{t}$ effective thickness

t thickness

T temperature

ρ_c char density

ρ_v uncharred material density

τ shear

ABLATOR EXPERIENCE - PRE-SHUTTLE

During the past 25 years a sound technology base has been developed for ablative materials in TPS applications. Extensive ground-based and flight experiments have led to the development of efficient, reliable, and predictable ablative TPS. Flight experience includes service on some notable space vehicles, both manned and unmanned, subjected to a wide range of environmental conditions (fig. 1). Without exception, the ablative heat shields on these vehicles performed satisfactorily and as predicted, despite the complexity of the ablative process.

Flight experience has demonstrated that a number of the analytical models developed by private and government laboratories (refs. 30-33) are capable of accurately predicting the behavior of ablative materials of known properties in a variety of environments. The analysis of reference 31, for example, was used to predict the performance of the Apollo 4 ablative TPS. As shown in figure 2, the calculated and measured surface recession and internal temperatures were in good agreement. Thus, at the beginning of the Shuttle Technology Program, ablative materials and analyses for predicting the behavior of these materials had been developed to an advanced state and had been used successfully to design ablative TPS for a number of manned and unmanned space vehicles.

SHUTTLE ABLATOR TECHNOLOGY PROGRAM

A program to develop an ablative TPS for shuttle was included in the Space Shuttle Technology Program from the beginning. An overview of the program for ablative TPS is shown in figure 3. This program was able to draw upon an extensive background of technology and flight experience while focusing on three major areas: materials, design and refurbishment. The specific questions that were addressed in the program and the reports addressing these questions were:

1. What ablation materials are suitable ? (Reference 20)
2. What defects are critical to the performance of an ablative TPS ? (References 25 and 27)
3. Can fabrication costs be reduced ? (Refs. 1-5, 14-19, and 23)
4. How would an ablative TPS be refurbished ? (Refs. 11-13 and 21)
5. What is the lowest weight, lowest cost, most efficient ablative TPS design ? (Refs. 24, 26 and 29)

As a result of these studies and previous experience, the advantages of using ablative TPS on the current shuttle orbiter's are:

1. Ablators have been proven reliable TPS.
2. Ablators are well characterized thermally and good analytical models are available.
3. Good candidate materials are available.
4. Refurbishment techniques have been developed for both direct bond and mechanically attached ablative panels.
5. Ablators are defect tolerant materials.
6. Excursions in the entry thermal environment are not catastropic.
7. Strain isolator pads are not required.
8. Ablator TPS design study demonstrated simple, direct bond application of large panels.

UNRESOLVED TECHNICAL ISSUES

The Shuttle Technology Program resulted in bringing the ablative TPS to a high degree of readiness for shuttle application. However, a few problem areas identified during the technology program remain to be solved and some problems may arise when ablator and RSI tiles are mixed in the same TPS. The following items, therefore, should be addressed to bring ablator technology up to current shuttle application readiness:

1. Material property design data base - satisfactory TPS design depends upon the availability of reliable thermophysical and mechanical property data. Some material poroperty data are available for most of the materials that might be considered for shuttle, but the statistical basis for the data is uncertain. Lack of sufficient mechanical property data is of particular concern.
2. Moisture-absorption - the low density ablation materials are generally hygroscopic. Moisture pick up could result in spalling of the material due to rapid outgassing, as well as increase in weight of the total heat shield. (See reference 24).
3. Sensitivity of ablator to sequential heat pulses - tests during the development of an ablative leading edge for shuttle, ref. 26, showed that excessive surface recession resulted when the ablative leading edge was sequentially exposed to ascent and entry heating pulses. Although similar performance may not be expected for the ablator tiles applied to the large areas of the shuttle because of lower temperatures and parallel flow, the performance of the ablator tiles subjected to sequential heating must be verified.
4. RSI/Ablator joint design - a number of studies have been conducted to investigate aerodynamic heating and erosion characteristics in heat shield systems with ablator/ablator

and ablator/RSI joints (refs. 24, 26, an 29). However, each of these studies had some limitations or deficiences and uncertainties remain in the expected performance of systems with regard to general shuttle applications. The general recommendations from these studies were that more work should be done on the concept of self-sealing gaps (un-filled) and that some additional effort should be expanded to find a low conductivity gap filler for these areas requiring a filler. Thus, the joint designs must be reexamined.

5. Induced stresses - the thermal and load induced stresses in an ablator tile need to be evaluated for possible unacceptably high stresses. The stresses induced in the RSI tile by adjacent ablator tiles also need to be evaluated.
6. Shock impingement on ablator tiles - excursions in the thermal and load environments are generally not catastropic for a honeycomb reinforced ablator. However, the strength of any aerodynamic shock waves that may impinge on the ablator tiles should be examined and the possible deliterious effects on the performance of ablator should be evaluated.
7. Ablator tile attachment and removal procedures - these were addressed in the technology programs and are discussed in references 9, 12, 13, and 24. However, ablator attachment and removal techniques and procedures must be reevaluated and made compatible with the latest shuttle systems designs and operations.
8. Contamination of RSI by ablation products and RTV outgas - one way the ablative TPS accommodates heat is by chemical degradation. The gaseous products from this degradation can and do deposit on RSI tiles. This deposition can be seen in figure 4. The RTV bond that might be used to apply ablative tiles may also outgas. This latter outgassing should be no worse than that experienced with the RSI tile application

methods currently used. The effects of the outgassing and deposition of ablation products on the shuttle orbit missions as well as the performance of the RSI tiles should be determined.

The NASA-Langley Research Center has initiated a task to examine and possibly solve some of these problem areas. This task is discussed in the next section.

NASA-LANGLEY RESEARCH CENTER TASK

The objective of the NASA-LaRC task is to identify candidate ablation materials for application to the shuttle orbiter TPS, assemble and assess the data base for these materials, and resolve key technical issues. The key issues that LaRC will address are the first five problem areas listed in the previous section, that is: (1) evaluate the adequacy of thermal and mechanical property data base, (2) evaluate moisture absorption, (3) determine the stresses in the ablator and stresses induced in the RSI by the ablator, (4) evaluate RSI/ablator tile joint designs, and (5) determine the sensitivity of the ablator to sequential heat pulses. The other three problem areas previously cited also need to be considered, but they may best be addressed by the Shuttle Project Office and/or other NASA Centers. The approach NASA-LaRC will take to meet the task objective will, in general, consist of a combination of short and highly focused experimental and analytical studies.

Recommended Ablators for Shuttle TPS

The aerodynamic heating rate histories, the aerodynamic total pressure and shear histories at four body points, figure 5, for the design trajectory (14414.1C), figure 6, and the nominal trajectory (STS-1), figure 7, for the shuttle orbiter have been examined. Based on these environments and previous flight vehicle experience and performance, two ablation materials are recommended as good candidates for shuttle application. One of the materials is the Viking heat shield material designated by the Martin Marietta Corporation, as SLA-561. This material is a silicone elastomeric resin filled with cork, hollow silica microspheres, hollow phenolic microspheres and chopped

silica fibers. The SLA-561 material performed well in the Viking program, has been extensively studied analytically and experimentally and should be well characterized. The SLA-561 has a density of about 14.5 lbs/ft^3 when reinforced with a phenolic glass honeycomb. The ablative heat shield design for shuttle reported in ref. 24 used the SLA-561 for the primary TPS material.

For areas on the orbiter where the recession of the SLA-561 ablator may be unacceptably high, the PRIME vehicle heat shield material is recommended. This material is designated by Martin Marietta Corporation as ESA-3560. This material is a filled silicone elastomeric composition with a density of 30 lbs/ft^3 when in a honeycomb reinforcement. The ESA-3560 ablator has also been extensively studied experimentally and analytically and should be well characterized.

<u>Material availability</u> - The Martin Marietta Corp. has advertised the use of both the SLA-561 and ESA-3560 as honeycomb reinforced thermal protection materials as recently as 1977. However, recent efforts to obtain specimens of these materials were restricted by the lack of the type of honeycomb used on the Viking and PRIME vehicles. The Martin Marietta Corp. has not produced any of this honeycomb since the Viking Program. If other honeycomb is used for these materials, the existing data base may not apply. Therefore, either a source for the Viking honeycomb must be found or a data base will have to be defined for these ablation materials in another honeycomb system.

<u>Predicted thermal performance</u> - Preliminary estimates of the thermal performance of ablator tiles at four different body points on the orbiter, figure 5, have been made. An implicit formulation of the analysis in reference 31 was used to calculate surface rescession and the back-surface temperature histories during entry. The model of the ablator tile is shown in figure 8. For these calculations, an RSI tile (not including the SIP) was assumed to be replaced by a tile of SLA 561 of equal thickness, which was bonded directly to the aluminum substructure. The body points considered, the corresponding ablator (RSI) tile thicknesses, and the effective thermal thickness of the aluminum substructure are listed in table I.

In these calculations, the thermophysical properties shown in table II were used directly except for the pyrolysis kinetics. The pyrolysis kinetics expression given in table II was modified, according to references 36 and 37,

to fit the plane pyrolysis zone model of reference 31. This model has been used to successfully predict the thermal response of a number of ablative materials (ref. 37). For the plane pyrolysis zone, the pyrolysis rate is given by $m_p = A \ \exp(-B/T)$. The modified pyrolysis kinetics used in the present study were A = 3510 lbm/ft^2-s-atm and B = 18095°R.

No general mechanistic surface recession model is available for the SLA 561 in honeycomb. Therefore, the surface was assumed to be all carbonaceous and to recede by oxidation only. These assumptions cause uncertainties in the surface recession because the high silicon content of the material may result in different oxidation parameters, and possible melting of the silicon in some environments is not taken into account. Additional work is required to define a more realistic surface recession model for both of the recommended ablative materials.

The heating rate histories for the two trajectories, figures 6 and 7, indicate that transition from laminar to turbulent flow occurred at body points 1702, 1800 and 213. This transition was accounted for in the calculations by using a turbulent mass blocking coefficient equal to one-third of the laminar blocking coefficient.

Calculations were made for the design trajectory, figure 6 and the nominal trajectory, figure 7. The char thicknesses, and surface recessions for each body point is given in table I. The corresponding back surface temperature histories are shown in figures 9 and 10. These calculations show that recession, at all body points, was about 0.05 inch or less and that recession occurred only at body points 1030 and 213 for the design trajectory and only at body points 1030 for the nominal trajectory. In all cases, the ablator thicknesses used was sufficient to keep the back surface temperature, below 810°R design temperature.

Assessment of Data Base

The data necessary to perform ablation analyses and thermal stress analyses for space shuttle applications of SLA 561 and ESA 3560 ablation materials include density, emittance, specific heat, and thermal conductivity in the

uncharred and charred state, and the strength, modulus, Poisson's Ratio, thermal expansion coefficient, and pyrolysis kinetics, in the uncharred state. In addition to the basic properties data, an understanding of the statistical basis for the data is needed.

Representative data have been compiled for the thermophysical and pyrolysis properties of SLA 561 ablation material (see table II). These data are complete, but their statistical basis is not known. The mechanical-properties data that have been compiled for uncharred SLA 561 include ultimate tensile strength and strain and the coefficient of thermal expansion as a function of temperature (see tables III and IV). Again, the statistical basis of these data is not known. Data that have been compiled for ESA 3560 ablation material (see tables V and VI) are comparable to those or SLA 561. A single value of $51.5 \times 10^{-6}\ R^{-1}$ is used for the ESA 3560 thermal expansion coefficient over the range 360°R to 504°R. The weakest link in the data base needed to verify suitability of ablation materials for shuttle applications is the availability of mechanical properties data.

Calculations were made, using ref. 31, to evaluate some of the effects of uncertainty in the thermophysical properties on the ablator tile thermal performance. The ablator property value with the most uncertainty is the thermal conductivity of the char layer. Calculations, discussed in the previous section, made with reported values for the SLA 561 ablator, table II, and assuming the RSI tile is replaced with an ablator tile of equal thickness, show that the TPS back-surface temperature does not exceed the 810°R design temperature. Similar calculations were made with a char conductivity 20% higher than the reported values. These calculations, table VII(a), showed that the back-surface temperature at only one body point (1702) exceeded the design temperature and then by only about 10°R. When the ablator thickness was increased by the thickness of the 0.16 inch SIP layer, which is not needed for the ablator, the back-surface temperature at all body points considered did not exceed the design temperature, table VII(b). The high char conductivity values also resulted in a thicker char and slightly less recession than the reported char conductivity values.

As previously mentioned, a recession model for SLA 561 is not available. The model used in the present study was used to predict the performance of a SLA 561 tile in a simulated shuttle entry heating environment, ref. 39. The calculated recession was in satisfactory agreement with the measured net thickness change of the tile.

Collectively, these calculations indicate that an ablator tile could replace an RSI tile without the back-surface temperature exceeding the design temperature, while maintaining the TPS outer-mold-line. The back-surface temperature of an ablator tile equal in thickness to the RSI tile plus SIP layer would not exceed the design temperature even if the char conductibility was 20% higher than reported values.

Moisture Absorption

Recent preliminary test data show that the equilibrium moisture content, by weight, of molded SLA 561 at 580°R and 95% relative humidity is about 6%. The honeycomb reinforced material is expected to absorb about the same amount of water. The need for a waterproof coating for SLA 561 was identified in an early ablator TPS design study (ref. 24). However, the coating recommended in reference 24 is not available.

Two approaches have been made to moisture proof the SLA 561 ablator. In one approach, a thin coating of RTV 655 silicone resin, the base resin for the SLA 561 ablator, was applied. This coating did not reduce moisture pick up at 580°R and 95% relativbe humidity and was judged unsatisfactory.

In the other approach, a thin coat (about 0.1% by weight) of Scotch Guard[1] was added. The weight gain at 580°R and 95% relative humidity was reduced to about 4%. After a two hour simulated rain at room temperature, SLA 561 with Scotch Guard gained 1.5%, 1% less than SLA 561 without Scotch Guard. The rain test is perhaps the most relevant test conducted. Although these are a limited

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1. Registered trademark of 3M Company

number of tests, the Scoth Guard is recommended. Tests will be conducted to determine the effects of rapid depressurization of wet samples on material integrity.

Stress Calculations

Stresses in the ablator tile will be calculated using the aerodynamic loads and the finite element stress analyses currently used by NASA LaRC for the RSI tiles and temperature distributions calculated with reference 31. An attempt will be made to assess the stresses induced in RSI tiles by adjacent ablator tiles. The data required for these calculations will come from the data base for the ablator which is being assessed.

Ablator/RSI Tile Joint Design

Current LaRC plans are to test flat panel specimens with different joint filler materials in high temperature arc-jet simulations of ascent and entry heating environrment. The test specimens will be similar to those in figure 11. The test environments will simulate the heating at two body points along the bottom centerline of the shuttle orbiter. The joint filler will be judged on the thermal performance of filler material, such as expansion out of the joint, as well as the overall performance of the RSI/ablator joint specimen, i.e. surface recession and roughness and back surface temperature response.

Ablator Sensitivity to Sequential Heat Pulses

Although the heating environments in reference 26, which resulted in poor ablator performance in sequential heating tests, was more severe than that expected on the large areas of the orbiter, the stability of the ablator in sequential heat pulses must be evaluated. Specimens similar to those shown in figure 11, with and without ablator/RSI joints, will be sequentially tested in simulated ascent and entry heating environments. The performance of the specimens will be judged by the amount of total recession, roughness and back surace temperature response.

CONCLUDING REMARKS

A study has been made to identify ablation materials for preflight application to the shuttle orbiter, assemble data needed for engineering assessment, and identify and/or resolve key technical issues related to shuttle application. The work done on ablative thermal protection systems for shuttle under the NASA Shuttle Technology Program has been briefly reviewed.

This review showed that ablators are a proven and reliable TPS and that an ablative TPS design exists for shuttle application. Calculations show that an ablator tile could replace an RSI tile and perform thermally in a very satisfactory manner in spite of uncertainties in reported values of the thermophyiscal properties. This review also revealed several areas of concern that need to be addressed before application to the current shuttle orbiter.

The NASA-LaRC initiated a task to examine some of these areas of concern. This task included short time, highly focused analytical and experimental programs to: (1) identify candidate ablation materials; (2) assess the data base for these materials; (3) evaluate the need and kind of waterproof coating; (4) calculate thermal stresses in an ablator tile; (5) identify an acceptable ablator/RSI tile joint filler; and (6) assess the sensitivity of the ablator to sequential heat pulses. The work in the first three of these areas is essentially complete. Two ablation materials have been identified for use on the shuttle. The Viking heat shield material, SLA 561, and the PRIME heat shield material, ESA 3560, were both made by the Martin Marietta Corporation. The ESA 3560 material would be used where recession of the SLA 561 would be unacceptable. Preliminary calculations showed that replacing an RSI tile with an equal thickness of SLA 561 would not cause back surface temperature to exceed 810°R, nor was the surface recession large in the areas considered. The supply of this material reinforced with the honeycomb used in earlier programs is essentially non-existent. The supply problem must be solved so that the data previously generated can be used or a data base on these materials with a different reinforcement must be generated.

Attempts have been made to establish and assess a data base for both ablation materials. Data have been compiled for both thermophysical and

mechanical properties; however, their statistical basis is not known. Calculations show that even with a 20% higher char thermal conductivity than reported for SLA 561, the back-surface temperature would not exceed 810°R. A complete validated mechanistic surface recession model for these materials is needed for reliable flight predictions although recession predicted with a simple oxidation model agreed with the measured net change in the thickness of specimens tested in a simulated shuttle entry heating environment. The weakest link in the data base is the availability of mechanical property data for both materials.

Work at LaRC is continuing in the other areas of concern. Results from these programs will be reported as they become available.

APPENDIX A
CONVERSION OF U.S. CUSTOMARY UNITS TO SI UNITS

PHYSICAL QUANTITY	U.S. CUSTOMARY UNITS	CONVERSION FACTOR (*)	SI UNITS (**)
Density	lbm/ft^3	16.018463	kg/m^3
Enthalpy	Btu/lbm	2.32×10^3	J/kg
Heating Rate	Btu/ft^2-s	1.134893×10^4	W/m^2
Pressure	lbf/ft^2	47.88	N/m^2
Stress	lbf/in^2	6.895x10x3	N/m^2
Specific Heat	Btu/lbm-°R	4.18×10^3	J/kg-K
Temperature	°R	1.8	K
Thermal Conductivity	Btu/ft-s-°R	6.24×10^3	W/m-k
Thickness	in.	2.54×10^{-2}	m

* Multiply value given in U. S. Customary Units by Conversion factor to obtain equivalent value in SI unit

** Prefixes to indicate multiples of units are as follows:

Prefix	Multiple
centi (c)	10^{-2}
kilo (k)	10^3
mega (m)	10^6

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TABLE I.- TPS THICKNESSES AT DIFFERENT BODY POINTS FOR SLA 561 ABLATOR TILES

BODY POINT	(a) RSI, IN.	$\bar{t}$, IN.	SLA 561	14414.1C TRAJECTORY ENTRY		STS-1 TRAJECTORY ENTRY	
				CHAR, IN.	RECESSION, IN.	CHAR, IN.	RECESSION, IN
1030	3.26	0.241	3.26	1.21	0.04	1.19	0.02
1702	0.81	0.274	0.81	0.67	0.00	0.68	0.00
1800	1.00	0.278	1.00	0.76	0.00	0.76	0.00
213	3.66	0.134	3.66	1.18	0.05	1.07	0.00

(a) Does not include SIP layer.

TABLE II - THERMOPHYSICAL PROPERTIES FOR SLA 561

VIRGIN MATERIAL

Density (Ref. 34). . . 14.5 lbm/ft^3

Thermal Conductivity (ref. 34), Btu/ft/s°R

Temperature, °R	10^{-9} atm	1.3×10^{-3} atm	1 atm.
510	6.0×10^{-6}	7.1×10^{-6}	8.5×10^{-6}
560	6.1×10^{-6}	7.2×10^{-6}	9.0×10^{-6}
610	6.1×10^{-6}	7.4×10^{-6}	9.6×10^{-6}
660	6.2×10^{-6}	7.5×10^{-6}	10.1×10^{-6}
710	6.2×10^{-6}	7.6×10^{-6}	10.6×10^{-6}
760	6.2×10^{-6}	7.8×10^{-6}	11.2×10^{-6}
810	6.3×10^{-6}	7.9×10^{-6}	11.8×10^{-6}
860	6.3×10^{-6}	8.1×10^{-6}	12.3×10^{-6}

Specific Heat (ref. 34), Btu/lbm °R

Temperature, °R	C_p
310	0.211
410	0.250
510	0.275
610	0.289
710	0.299
810	0.301

Pyrolysis Kinetics (ref. 34)

$$\frac{d\rho}{dt} = (\rho_v - \rho_c)\ 2.78 \times 10^{10} \left(\frac{\rho - \rho_c}{\rho_v - \rho_c}\right)^3 \exp\left(\frac{-34200}{T}\right)$$

TABLE II (Continued

PYROLYSIS GASES

Heat of Pyrolysis. 0

Specific Heat, Btu/lbm °R. . . 0.6

CHARRED MATERIAL

Density (ref. 34) . . . . 7.98 lbm/ft^3

Emissivity (ref. 34) 0.9

Thermal Conductivity (ref. 35), Btu/ft-s °R

Temperature, °R	k
400	15.0×10^{-6}
1600	15.1×10^{-6}
1800	16.4×10^{-6}
2000	18.2×10^{-6}
2200	20.6×10^{-6}
2400	24.0×10^{-6}
2600	27.6×10^{-6}
2800	31.6×10^{-6}
3000	36.0×10^{-6}
3200	41.0×10^{-6}
3400	47.0×10^{-6}
3600	54.1×10^{-6}

Table II - (Concluded)

Specific Heat (ref. 34)

Temperature, °R	Btu/lbm °R C_p
600	0.195
800	0.231
1000	0.268
1200	0.297
1400	0.320
1600	0.343
1800	0.363
2000	0.383
2200	0.400
2400	0.413

Oxidation Kinetics (ref. 36)

Order of oxidation 1
Activation temperature . . . .76500 °R
Reaction rate constant . . . .10^{10} lbm/ft^2-s-atm

TABLE III. - MECHANICAL PROPERTIES DATA VERSUS TEMPERATURES FOR SLA 561 ABLATION MATERIAL

Temp., R	Ultimate Tensile Strength, psi	Ultimate Strain, in/in	Ultimate Secant Modulus, ksi
310	132	.0073	18.1
360	80	.0200	4.0
410	60	.0210	2.9
461	49	.0216	2.3
509	42	.0216	1.9
560	38	.0216	1.8
610	34	.0215	1.6
661	31	.0215	1.4
709	29	.0214	1.4
760	28	.0212	1.3
810	28	.0194	1.4
860	27	.0165	1.6
911	25	.0133	1.9
959	21	.0095	2.2
1010	16	.0050	3.2
1060	5	.0006	8.3

TABLE IV. - THERMAL EXPANSION COEFFICIENT VERSUS TEMPERATURE FOR SLA 561 ABLATION MATERIAL

Temperature R	Thermal Expansion Coefficient, 10^{-6}/R
40	18.4
572	14.0
860	15.5

TABLE V. - THERMOPHYSICAL PROPERTIES OF ESA 3560

VIRGIN MATERIAL

Density (ref. 38) ....30 lbm/ft^3

Thermal Conductivity (ref. 38), Btu/ft-s- °R

Temperature, °R	k
200	9.54×10^{-6}
300	11.16×10^{-6}
400	12.60×10^{-6}
500	15.60×10^{-6}
600	13.92×10^{-6}
700	15.00×10^{-6}
800	15.60×10^{-6}
900	15.84×10^{-6}

Specific Heat (ref. 38), Btu/lbm °R

Temperature, °R	C_p
100	0.032
200	0.100
300	0.162
400	0.222
500	0.280
600	0.330
700	0.370

Pyrolysis Kinetics (ref. 38)

$$\frac{d\rho}{dt} = (\rho_v - \rho_c)\ 4\times10^{10}\left(\frac{\rho - \rho_c}{\rho_v - \rho_c}\right)^2 \exp\left(\frac{-34100}{T}\right)$$

TABLE V. - (Continued)

PYROLYSIS GASES

Heat of Pyrolysis 0

Specific Heat, Btu/lbm °R........0.6

CHARRED MATERIAL

Density (ref. 38) ....22 lbm/ft^3

Emissivity...................... 0.9

Thermal Conductivity (ref. 38) Btu/ft-s-°R

Temperature, °R	k
1500	76.8×10^{-6}
2500	94.8×10^{-6}

Specific Heat (estimated) Btu/lbm °R

Temperature, °R	C_p
1400	0.195
1600	0.225
1800	0.245
2000	0.260
2200	0.270

TABLE VI. - MECHANICAL PROPERTIES DATA VERSUS TEMPERATURE FOR ESA 3560 ABLATION MATERIAL

Temp., °R	Ultimate Tensile Strength, psi	Ultimate Strain, in/in	Ultimate Modulus	
			Tangent, ksi	Secant ksi
211	2103	.008	235	262
259	1798	.010	180	180
310	537	.090	59	5.8
360	261	.220	16	1.5
410	174	.170	4.4	1.5
461	131	.130	4.4	1.5
509	116	.110	4.4	1.5
569	102	.090	4.4	1.5
610	102	.080	4.4	1.51
661	87	.060	4 4	1.5

TABLE VII - CALCULATED THICKNESSES AND TEMPERATURES FOR SLA 561 ABLATOR WITH 20% INCREASE IN CHAR THERMAL CONDUCTIVITY

(a) Ablator Tile Thickness Equals RSI Tile Thickness.

Trajectory	Body Point	Maximum (1) Back Surface Temperature °R	Initial Thickness in.	Reces-sion in.	Char Thick-ness in.	Uncharred Thickness in.
14414.1C	1030	595	3.26	0.038	1.325	1.897
	1702	818	0.81	0.000	0.709	0.101
	1800	739	1.00	0.000	0.809	0.190
	213	592	3.66	0.045	1.295	2.320
STS-1	1030	593	3.26	0.014	1.302	1.945
	1702	807	0.81	0.000	0.714	0.096
	1800	725	1.00	0.000	0.803	0.197
	213	584	3.66	0.000	1.165	2.495

(1) Initial temperature 560°R

TABLE VII - CALCULATED THICKNESSES AND TEMPERATURES FOR SLA 561 ABLATOR WITH 20% INCREASE IN CHAR THERMAL CONDUCTIVITY (CONTINUED)

(b) Ablator Tile Thickness Equals RSI Plus SIP Thickness.

Trajectory	Body Point	Maximum (1) Back Surface Temperature °R	Initial (2) Thickness in.	Recession in.	Char Thickness in.	Uncharred Thickness in.
14414.1C	1030	590	3.420	0.038	1.325	2.057
	1702	753	0.970	0.000	0.798	0.172
	1800	698	1.160	0.000	0.877	0.283
	213	587	3.820	0.045	1.295	2.479
STS-1	1030	588	3.420	0.014	1.301	2.105
	1702	745	0.970	0.000	0.804	0.166
	1800	688	1.160	0.000	0.864	0.296
	213	580	3.820	0.000	1.163	2.655

(1) Initial temperature 560°R
(2) RSI thickness plus 0.16 inch, SIP thickness

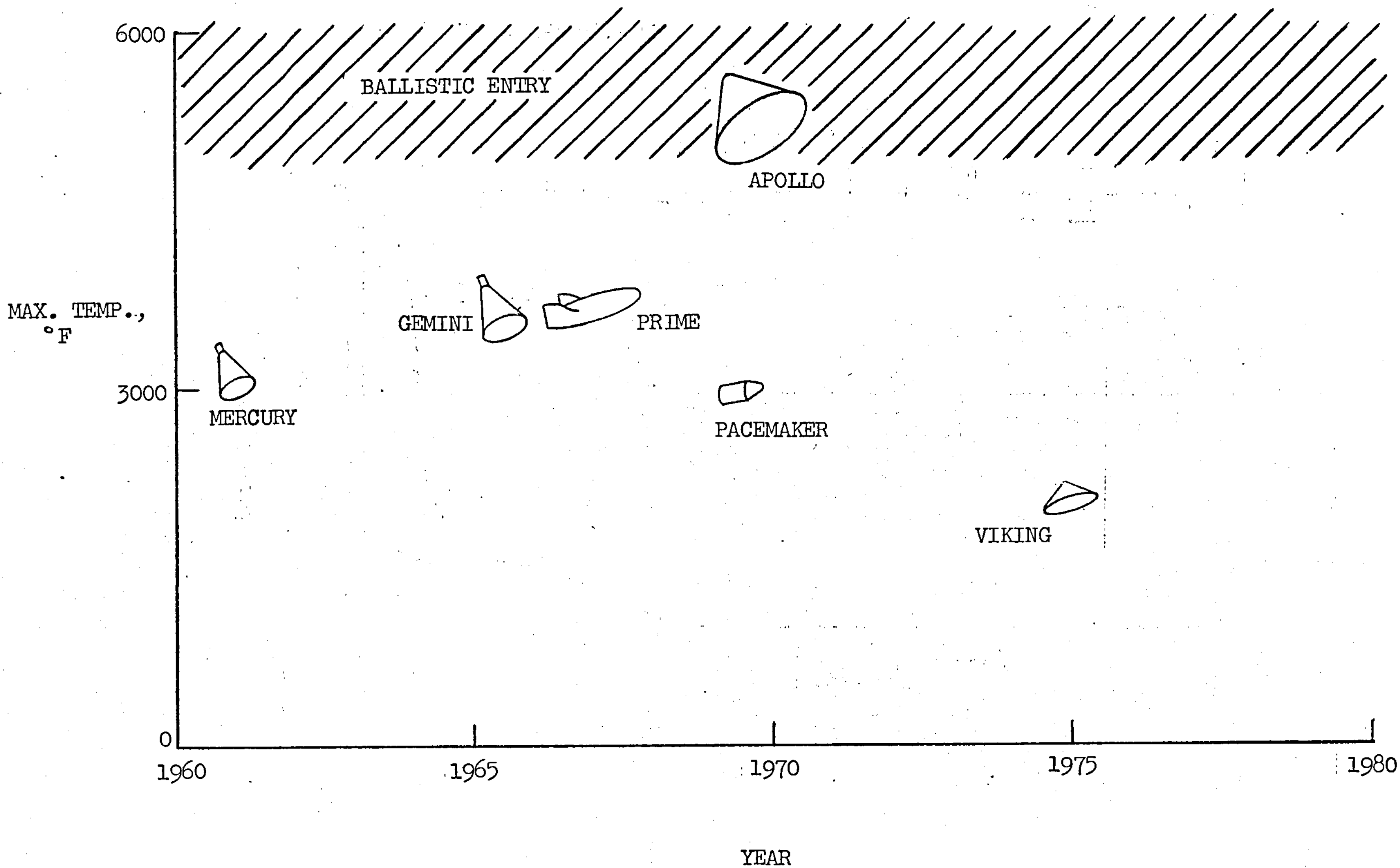
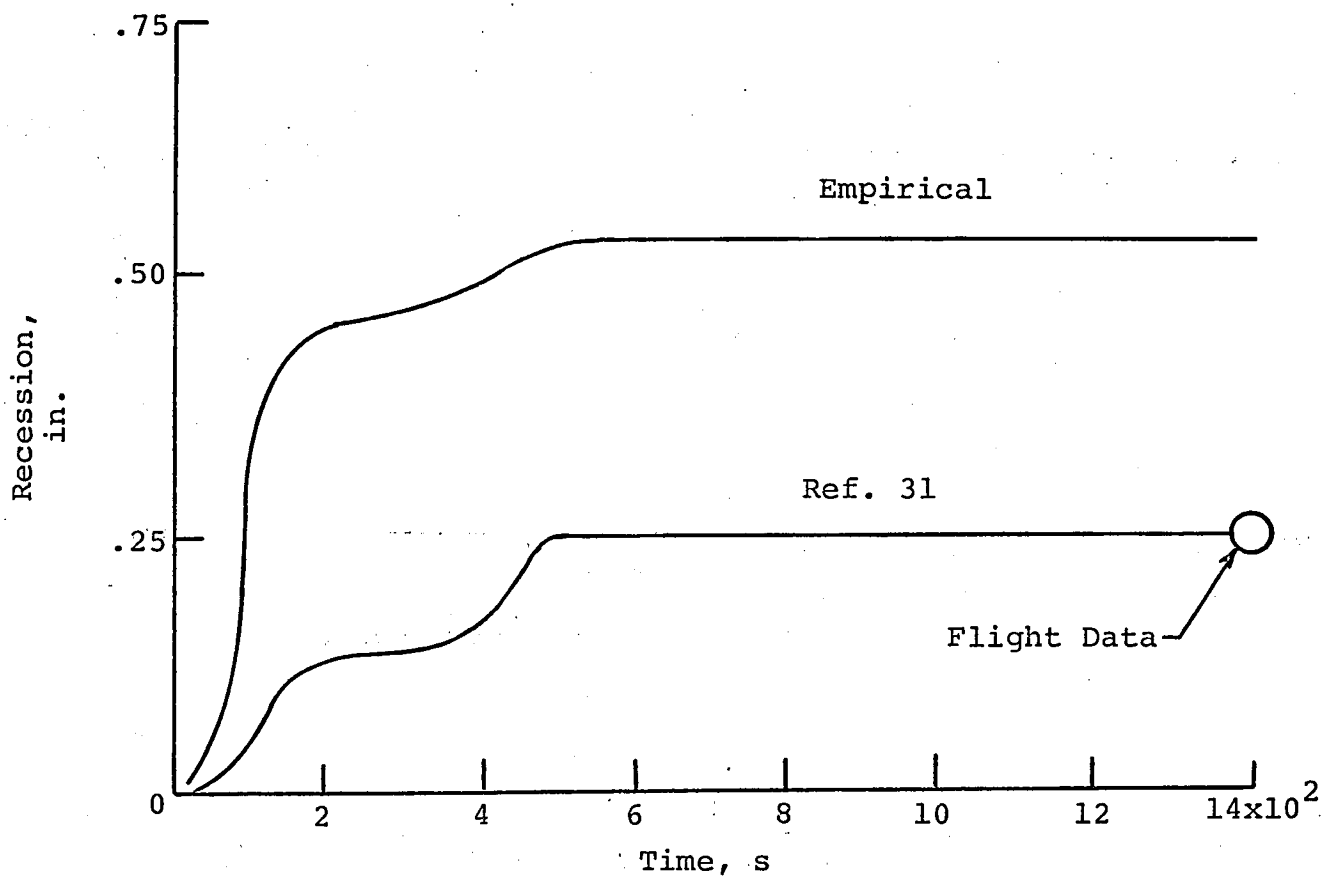
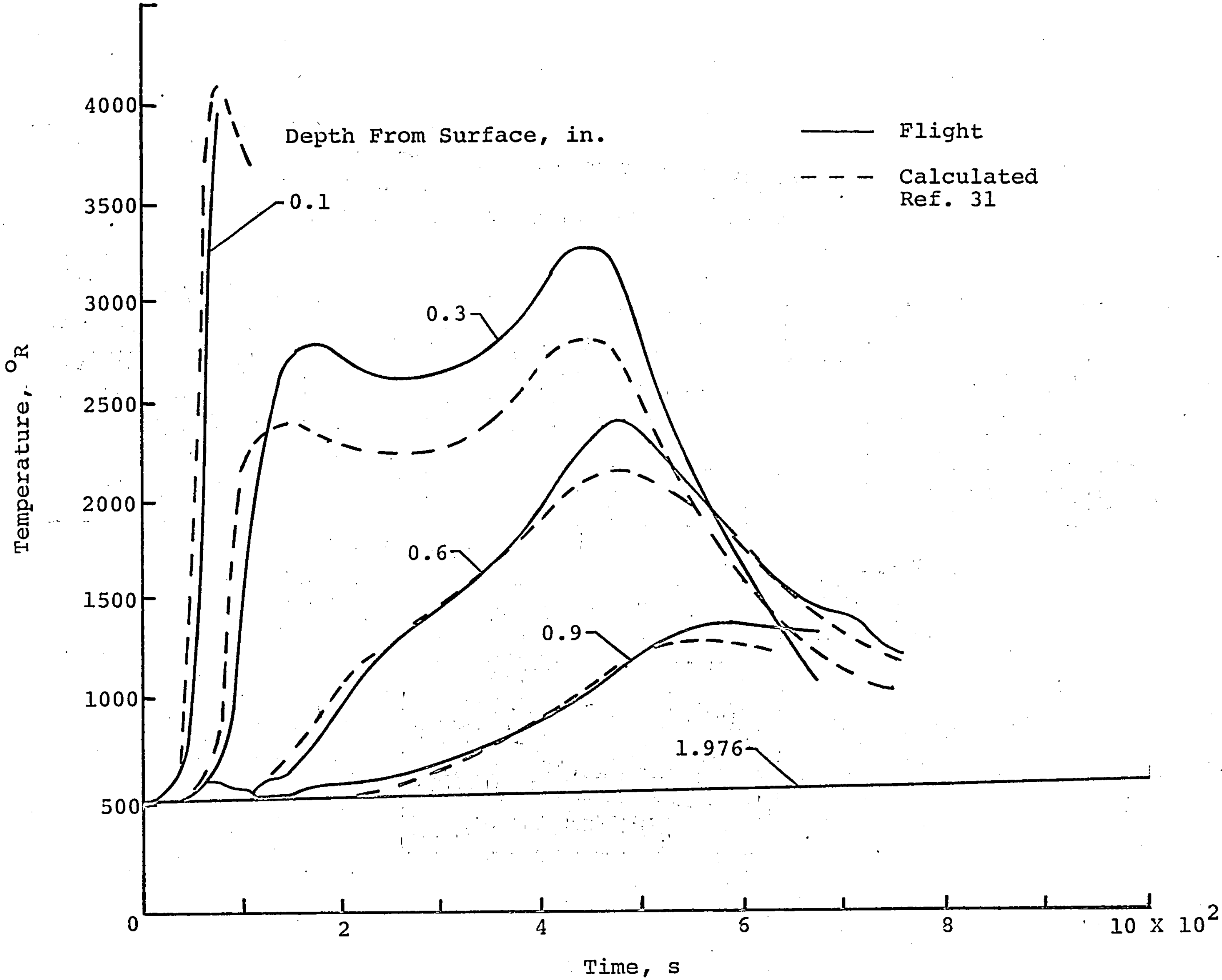


Figure 1.- Flight applications of ablative heat shields.



(a) Surface recession.

Figure 2 .- Postflight calculations of recession and temperatures for Apollo 4, body point 705.



(b) Internal temperatures.

Figure 2 - Concluded.

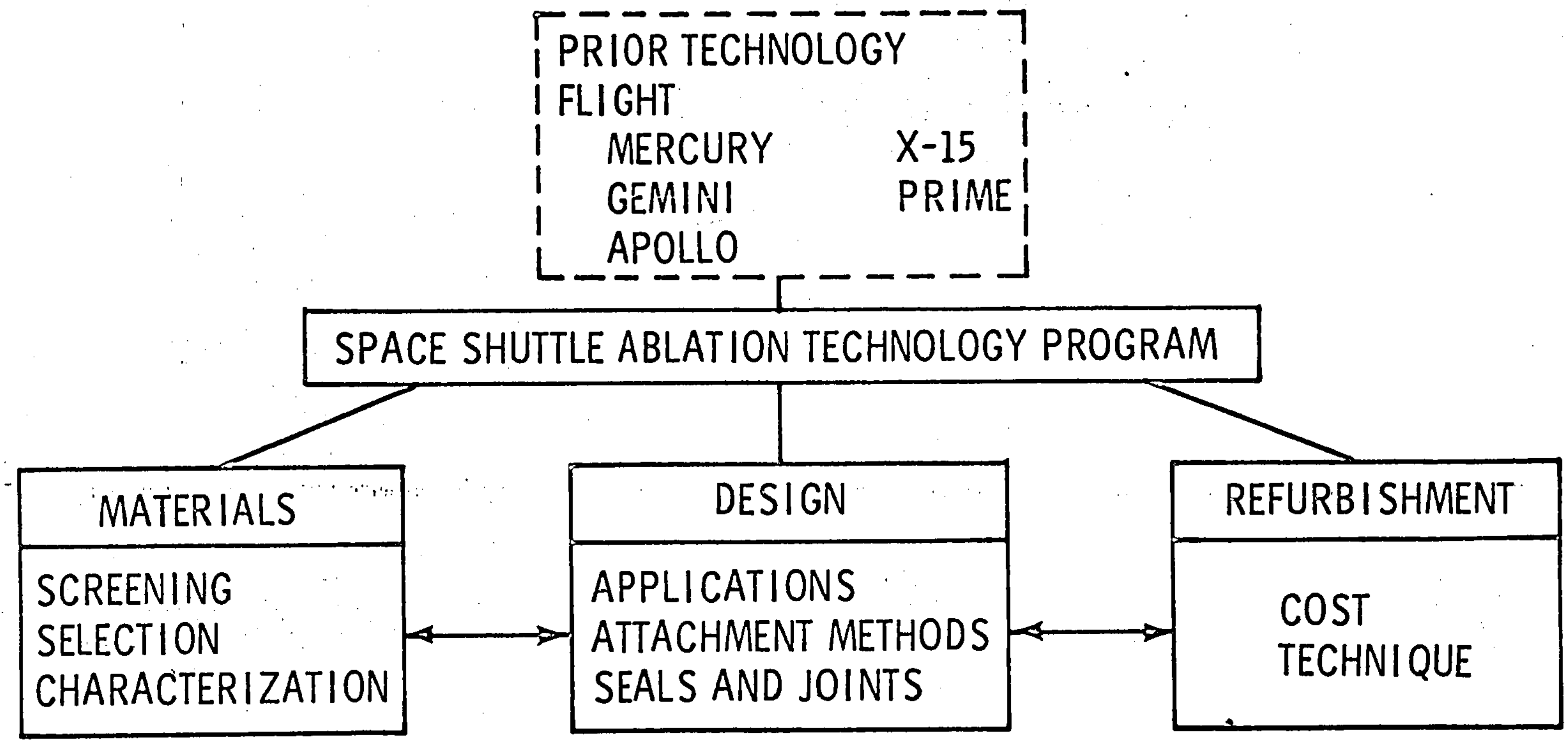


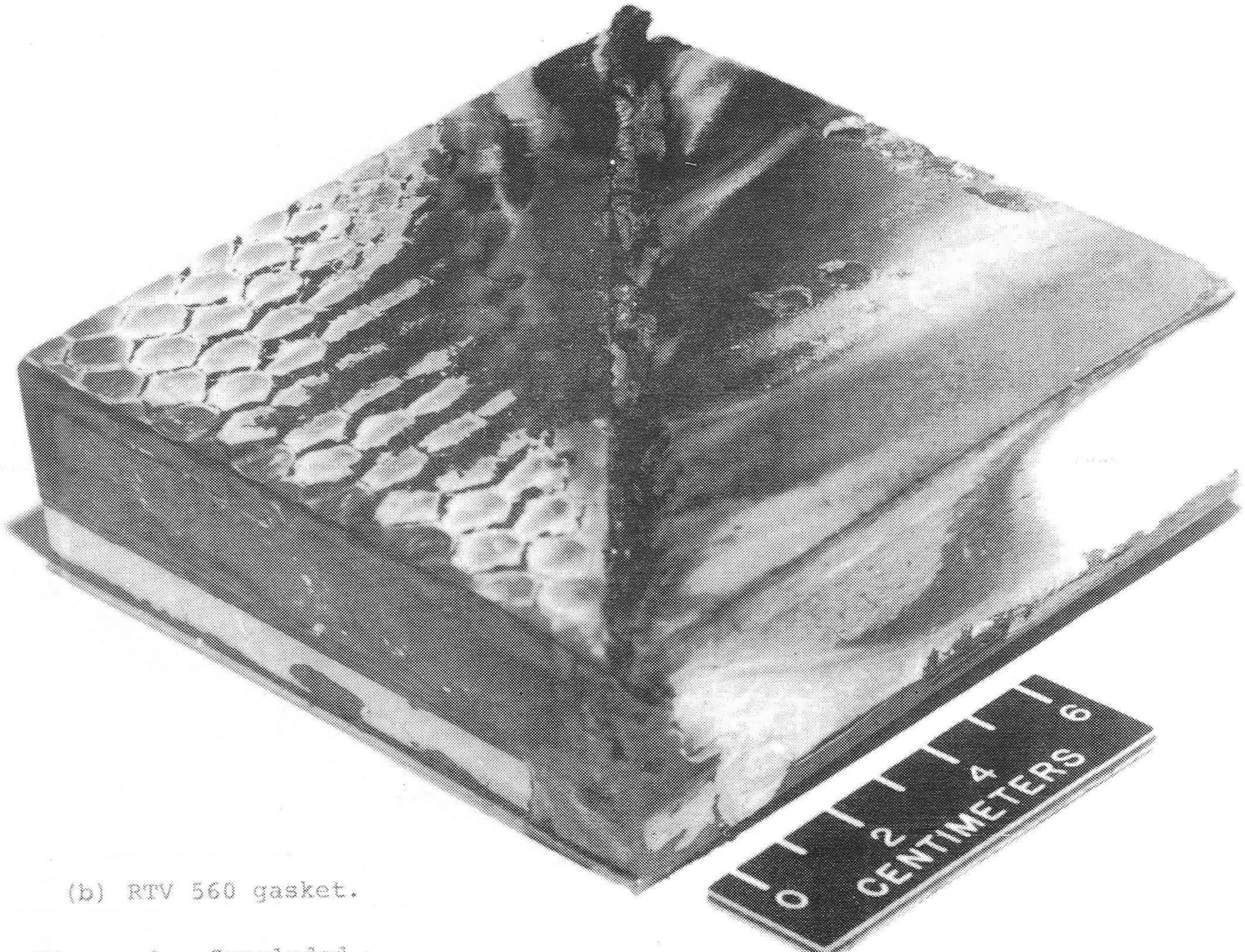
Figure 3.- Outline of ~~Space~~ Shuttle Ablative Heat Shield Technology Program.

NASA
L-74-4091

(a) Silica felt gasket.

Figure 4.- Ablator/RSI joint test models.

NASA
L-74-4093



(b) RTV 560 gasket.

Figure 4.- Concluded.

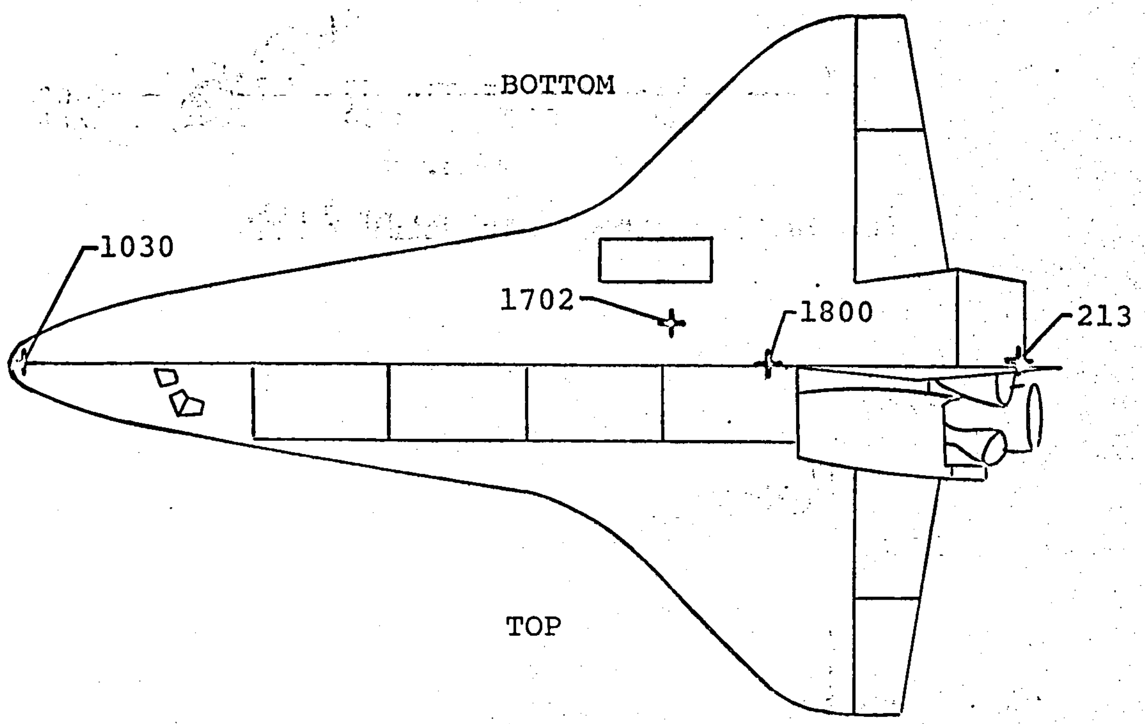
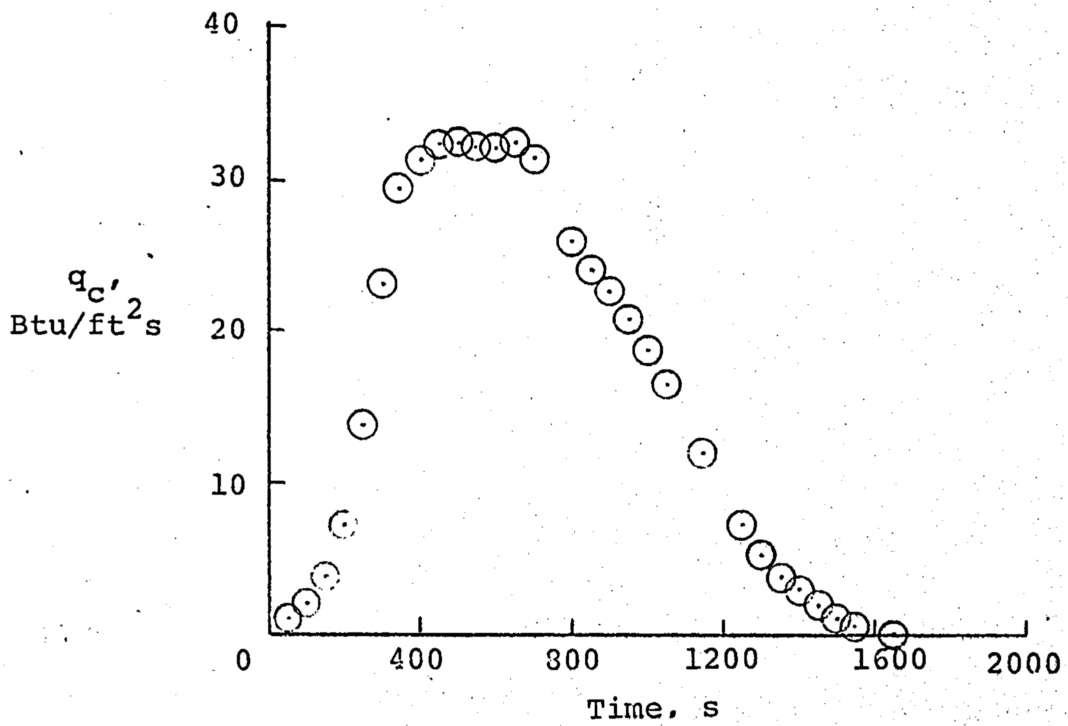
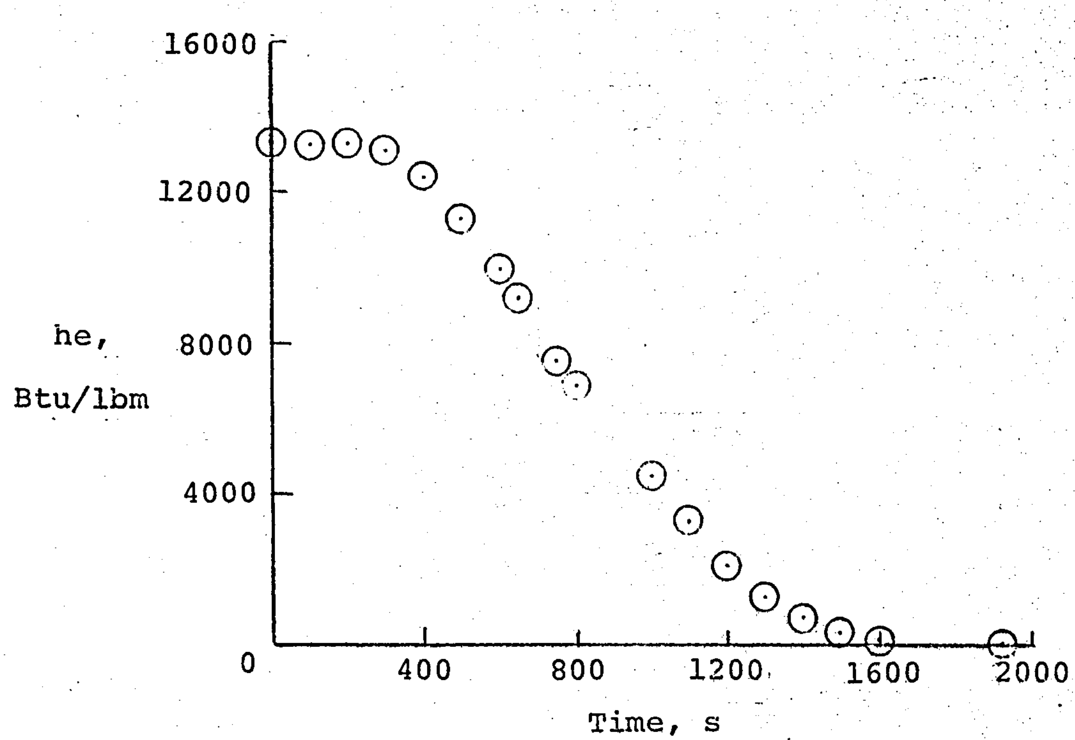


Figure 5.- Relative location of shuttle body points 1030, 1702, 1800 and 213.

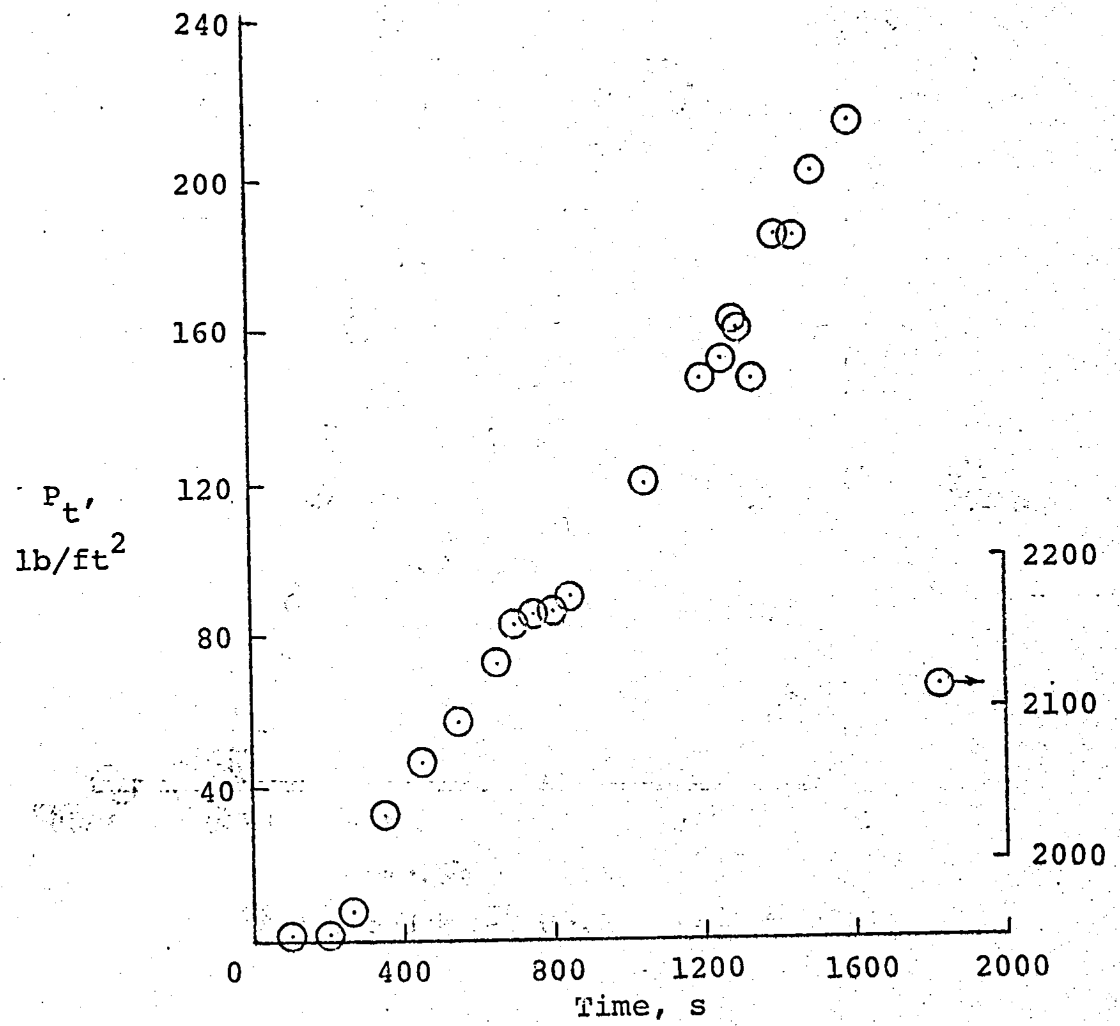


(a) Heating rate, body point 1030.

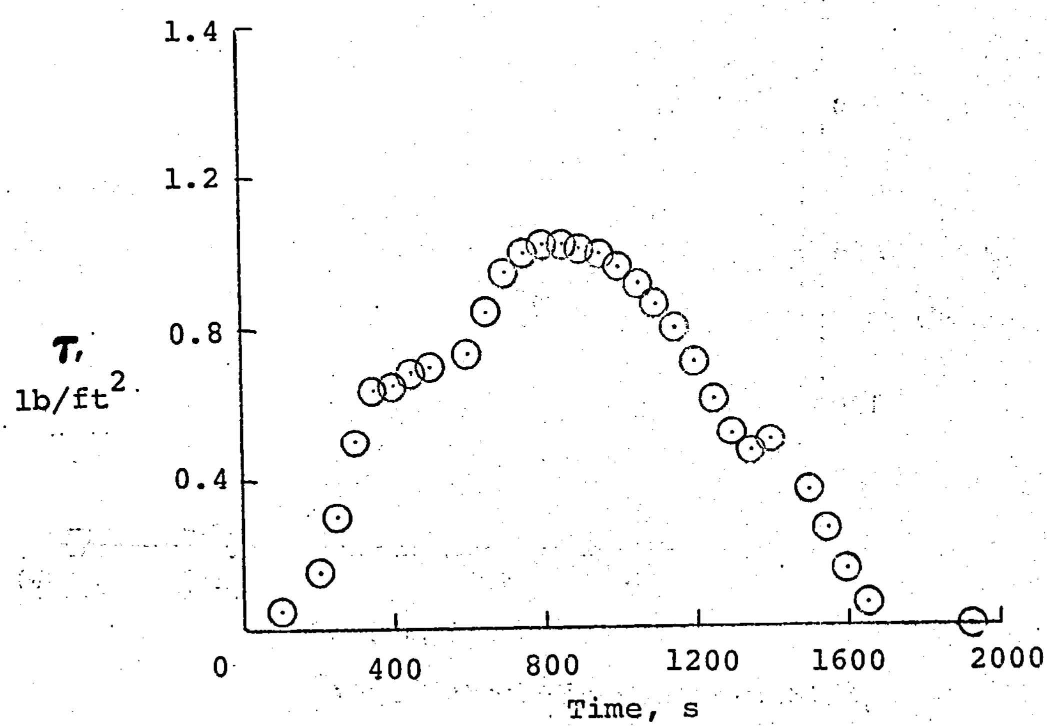


(b) Enthalpy, body point 1030.

Figure 6.- Predicted heating rate, enthalpy, pressure and shear for orbiter body points 1030, 1702, 1800 and 213 for the design entry trajectory 14414.1C.

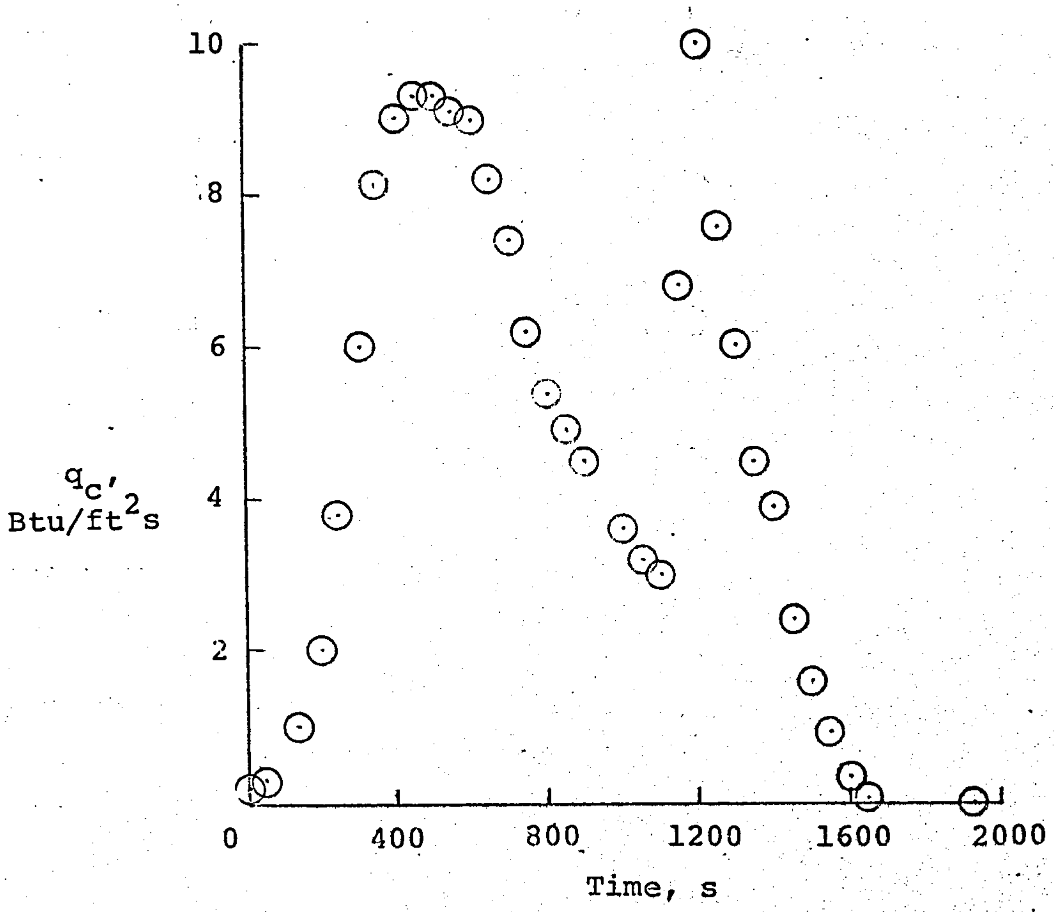


(c) Pressure, body point 1030.

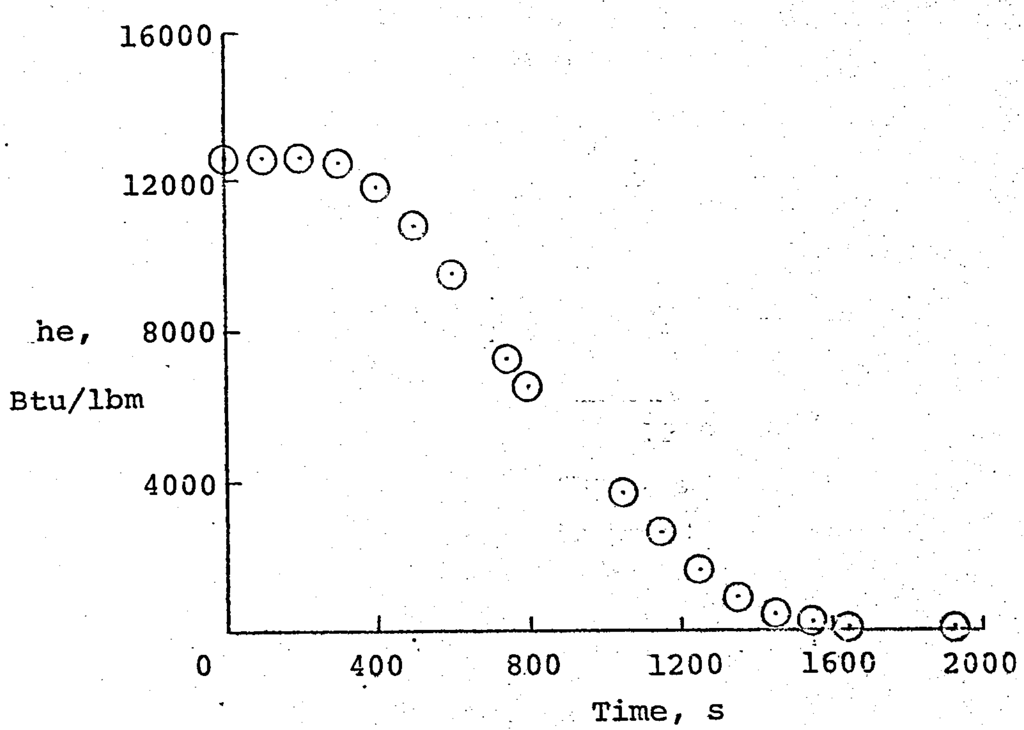


(d) Shear, body point 1030.

Figure 6.- Continued.

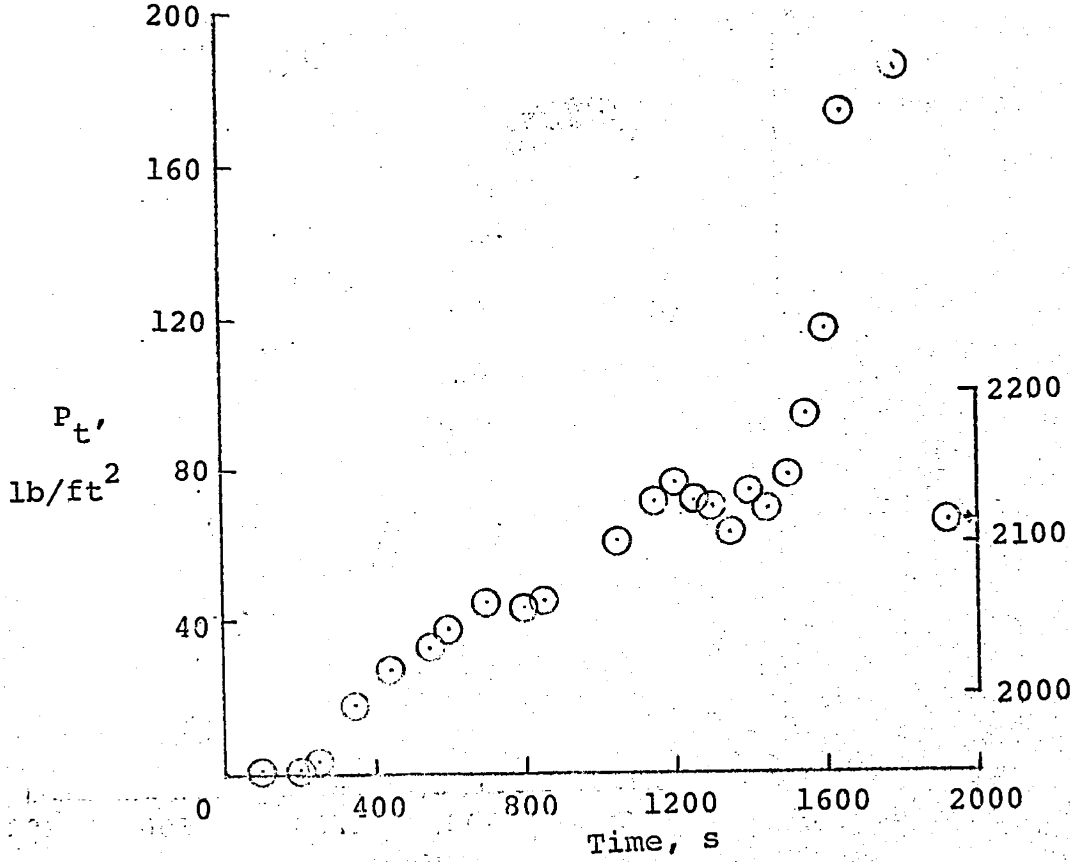


(e) Heating rate, body point 1702.

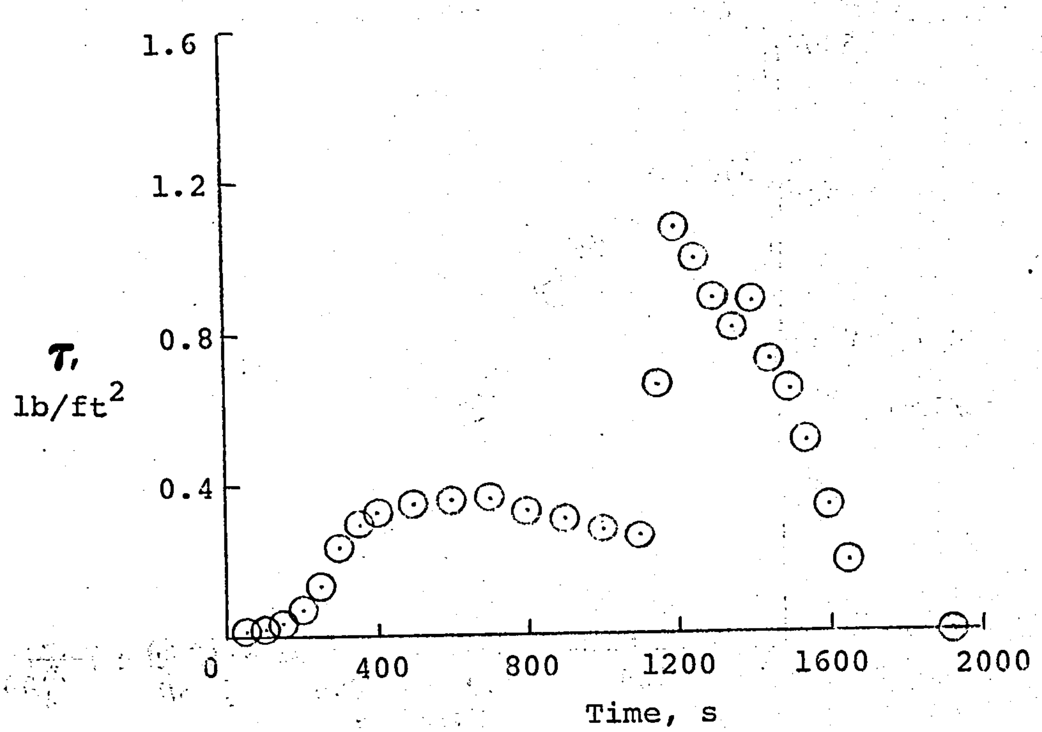


(f) Enthalpy, body point 1702.

Figure 6.- Continued.

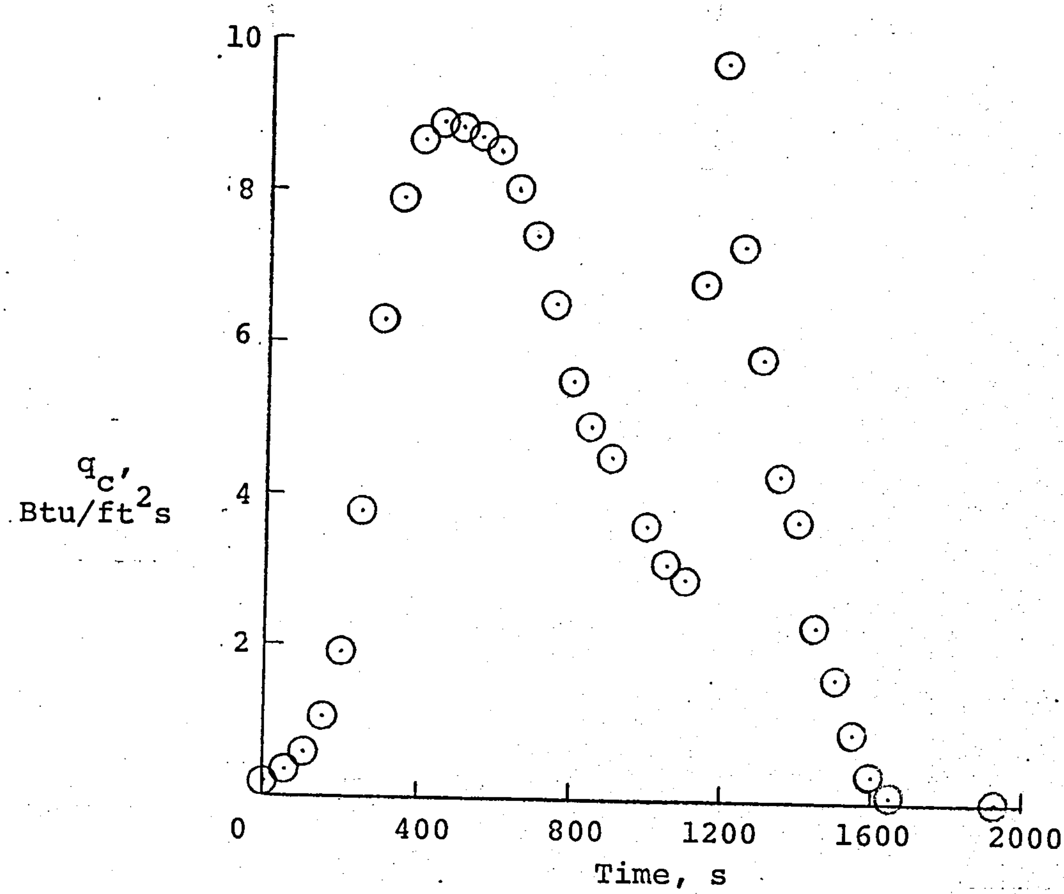


(g) Pressure, body point 1702.

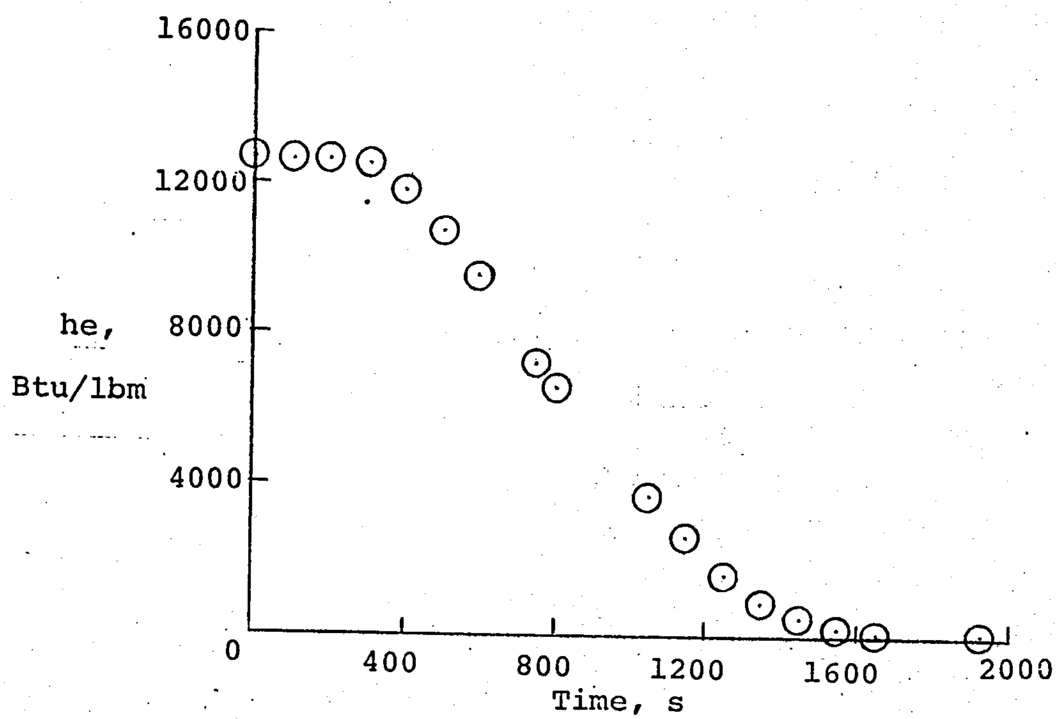


(h) Shear, body point 1702.

Figure 6.- Continued.

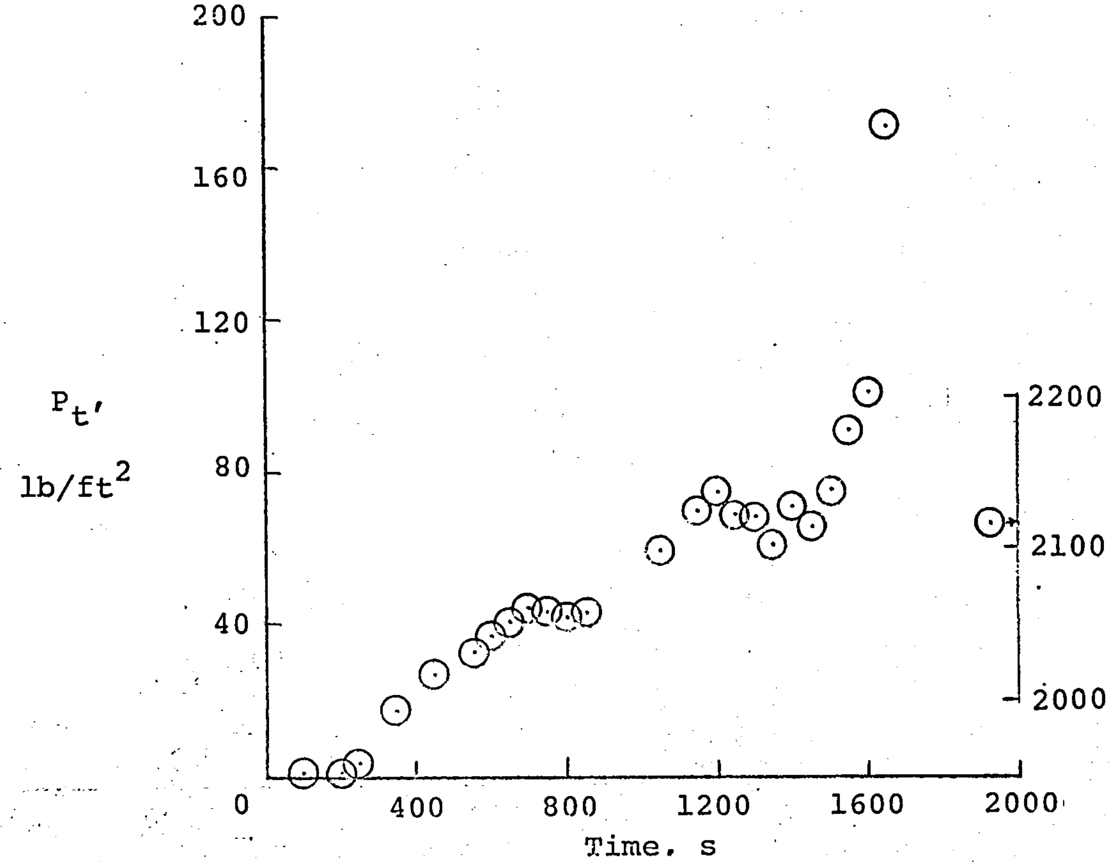


(i) Heating rate, body point 1800.

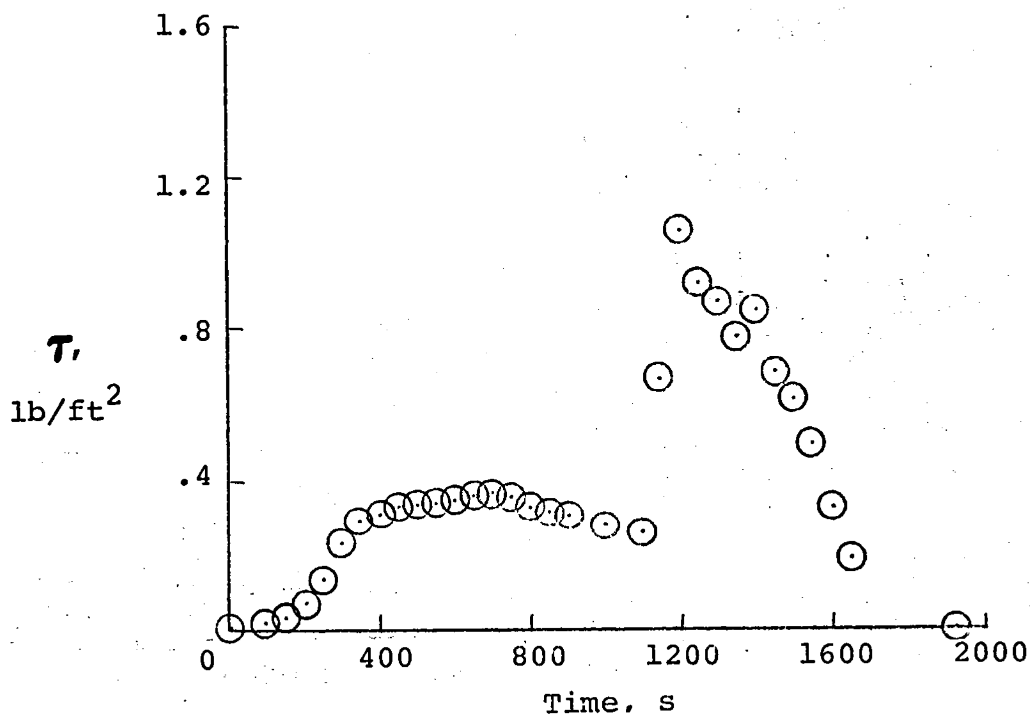


(j) Enthalpy, body point 1800.

Figure 6.- Continued.

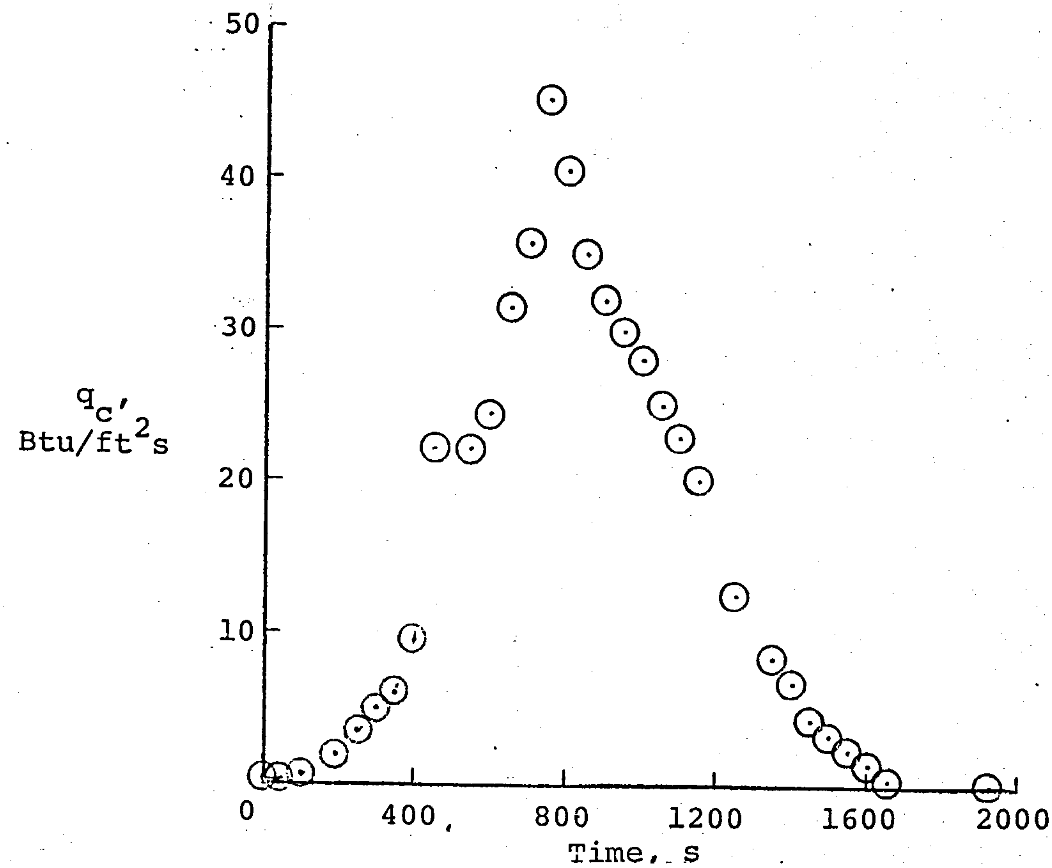


(k) Pressure, body point 1800.

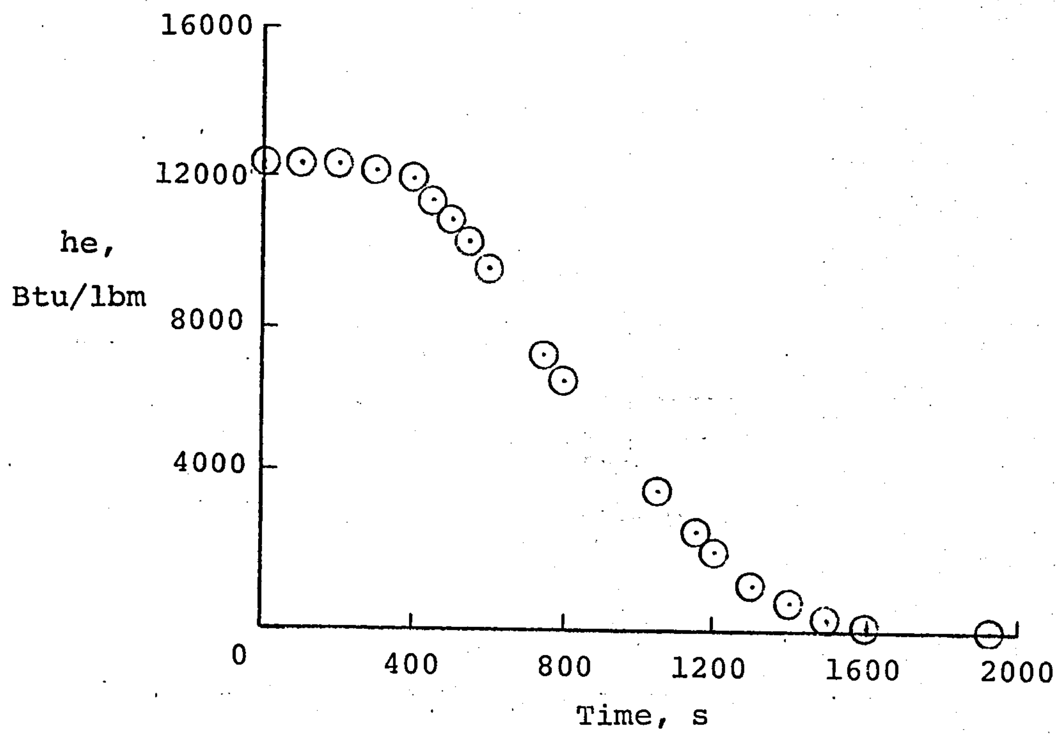


(l) Shear, body point 1800.

Figure 6.- Continued.

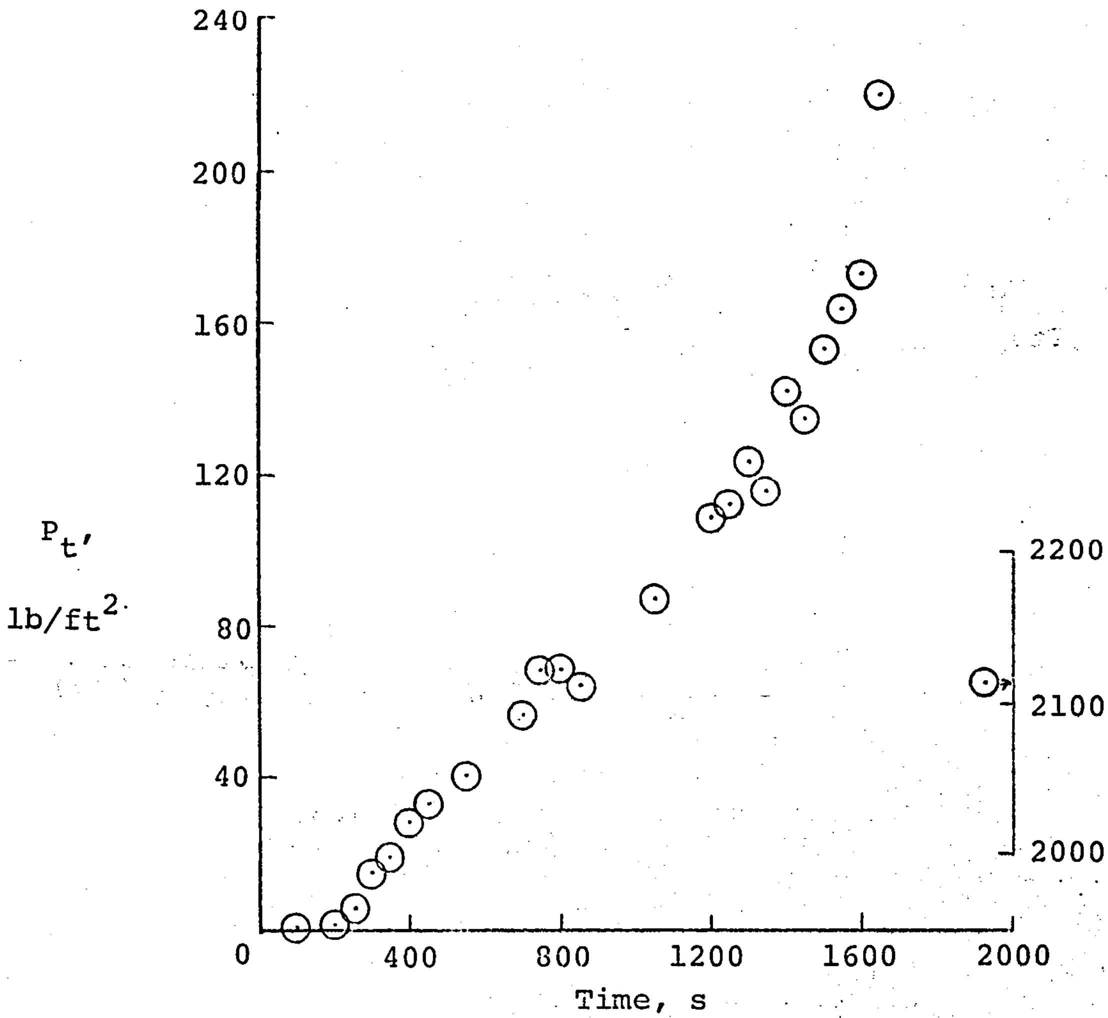


(m) Heating rate, body point 213.

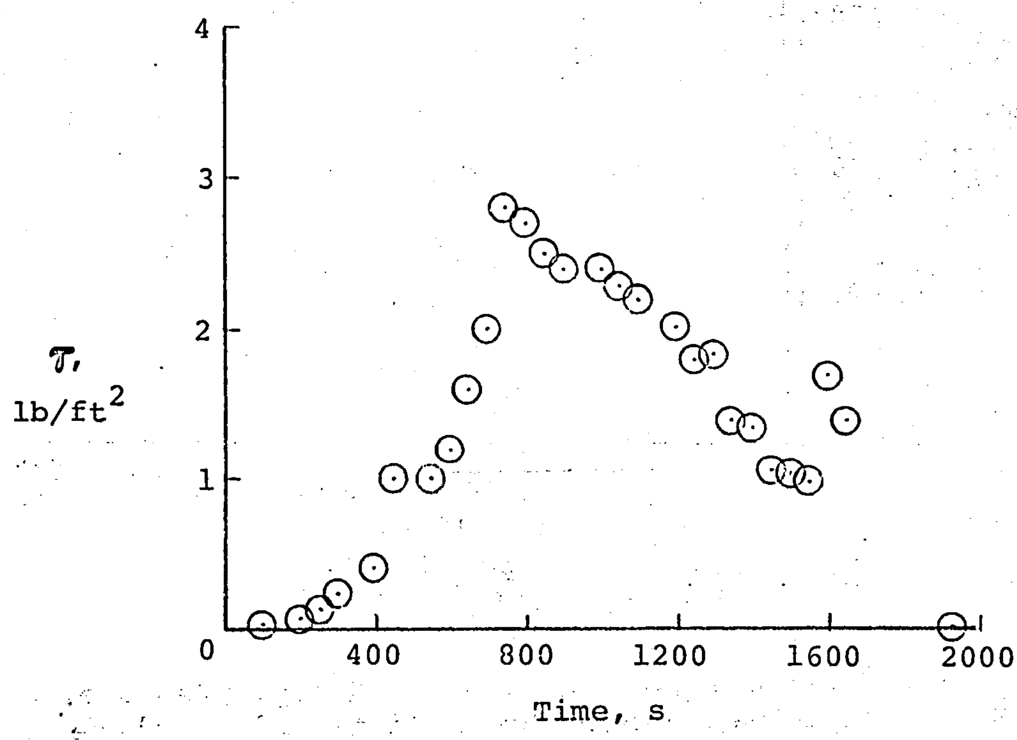


(n) Enthalpy, body point 213.

Figure 6.- Continued

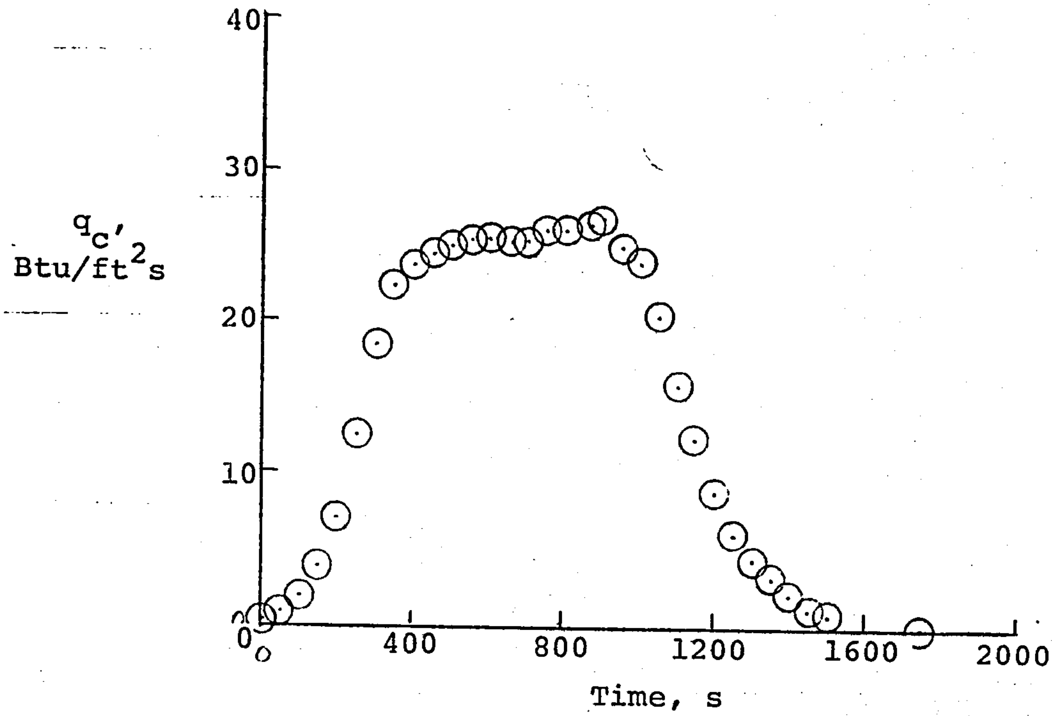


(o) Pressure, body point 213.

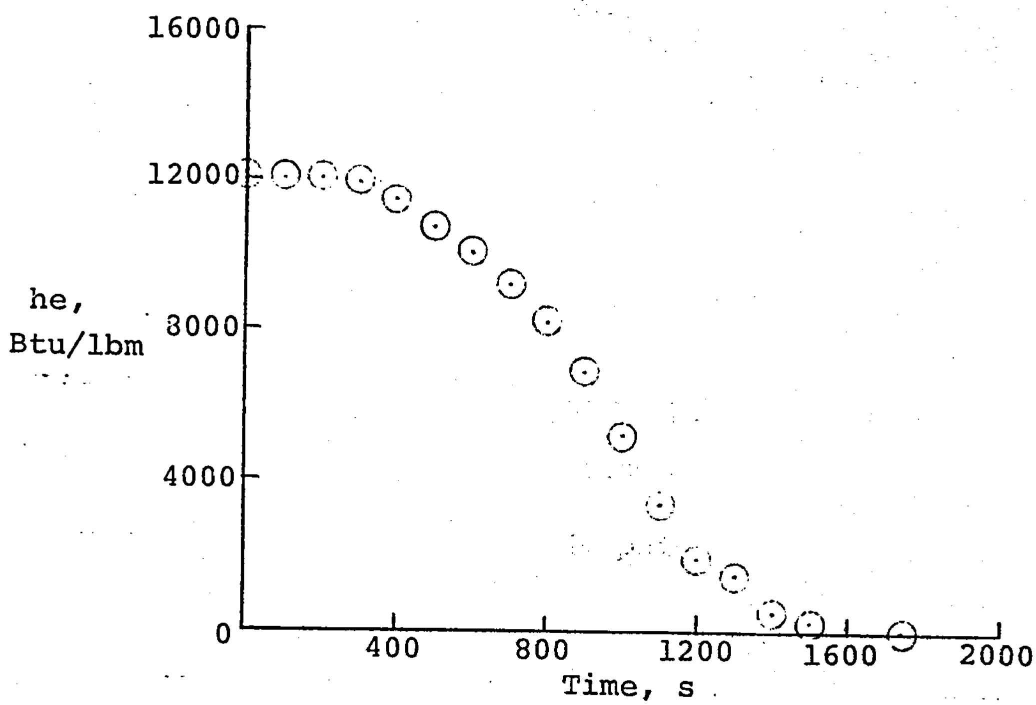


(p) Shear, body point 213.

Figure 6.- Concluded.

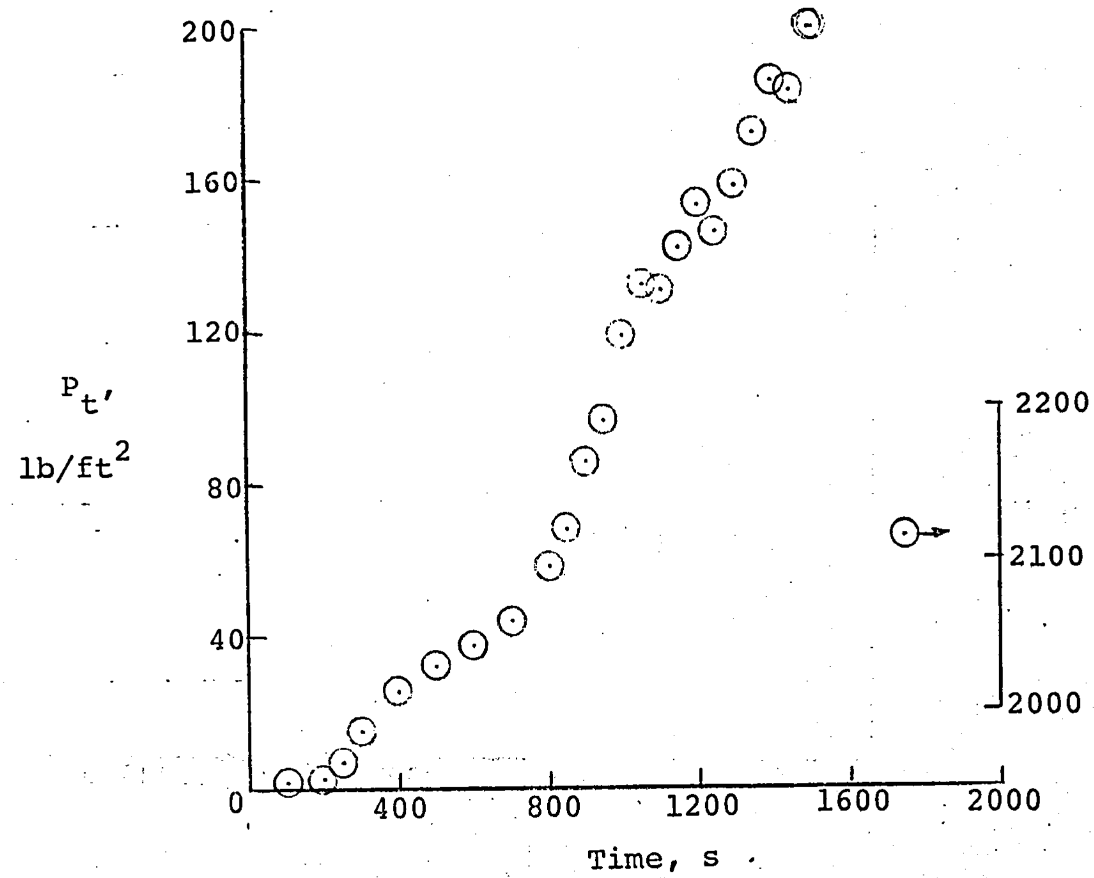


(a) Heating rate, body point 1030.

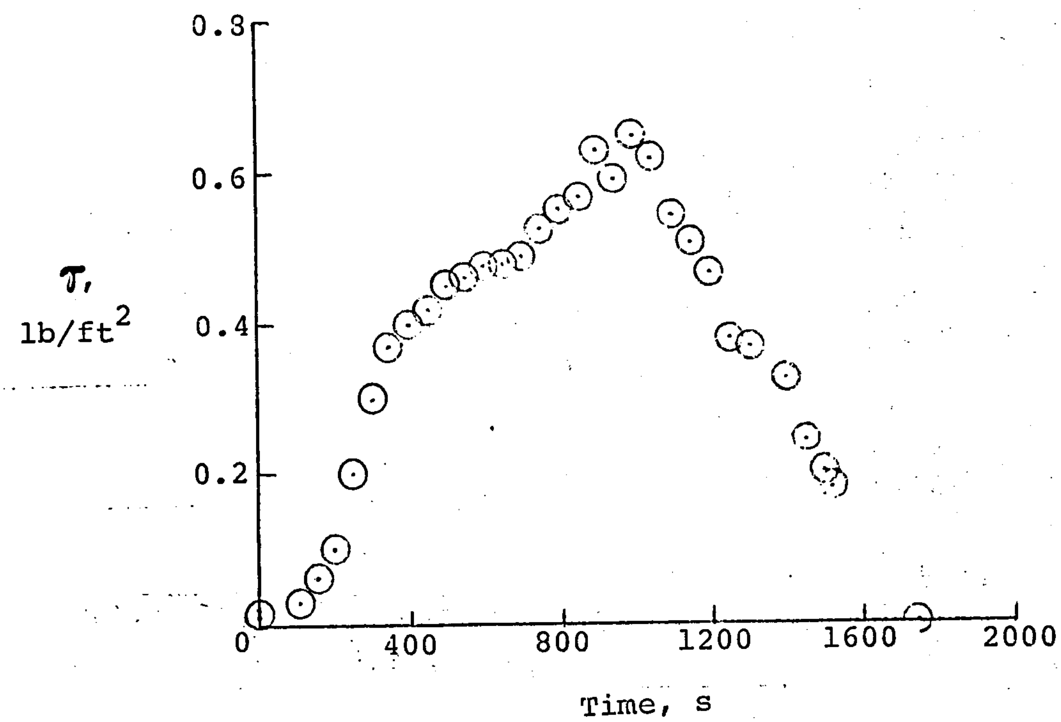


(b) Enthalpy, body point 1030.

Figure 7.- Predicted heating rate, enthalpy, pressure, and shear for orbiter body points 1030, 1702, 1800, and 213 for the nominal entry trajectory STS-1.

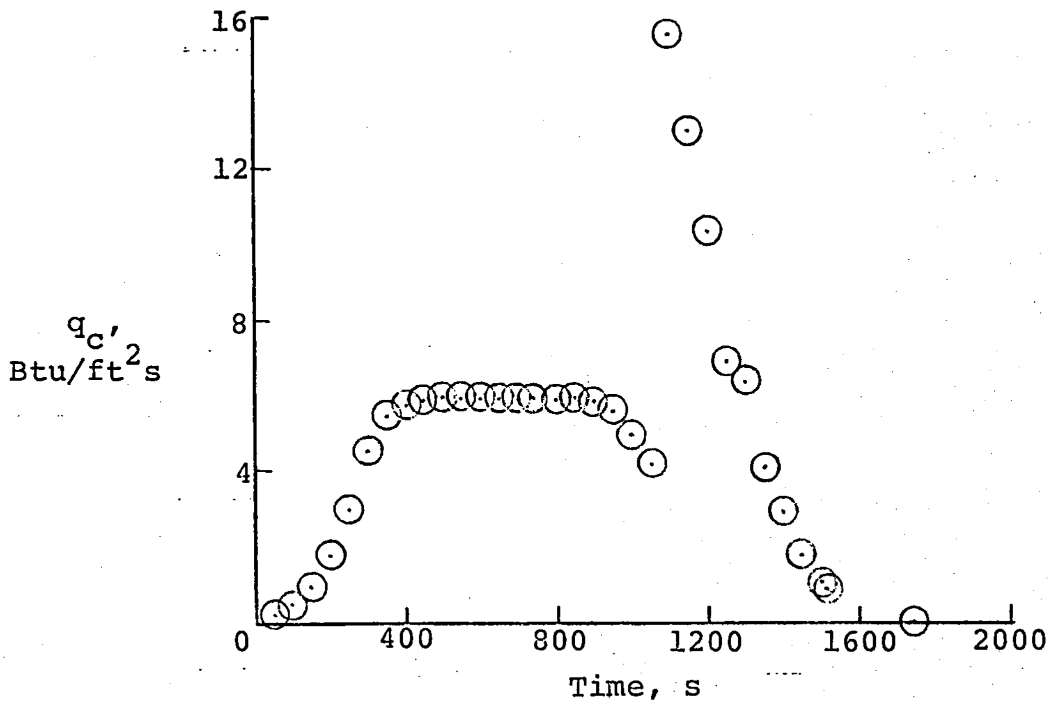


(c) Pressure, body point 1030.

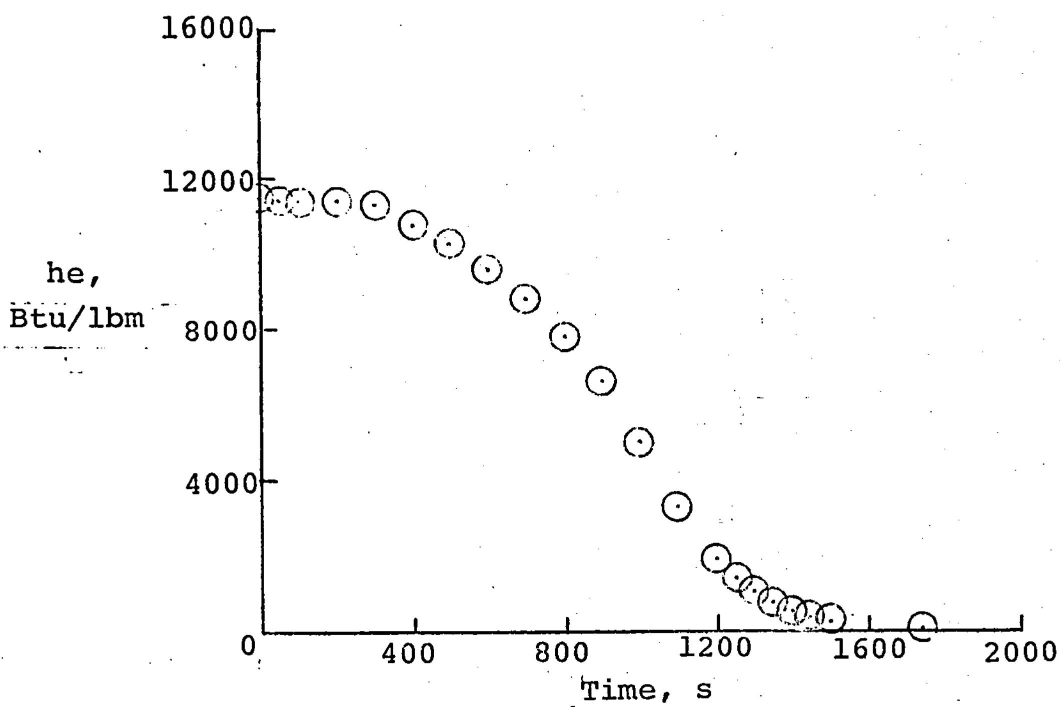


(d) Shear, body point 1030.

Figure 7.- Continued.

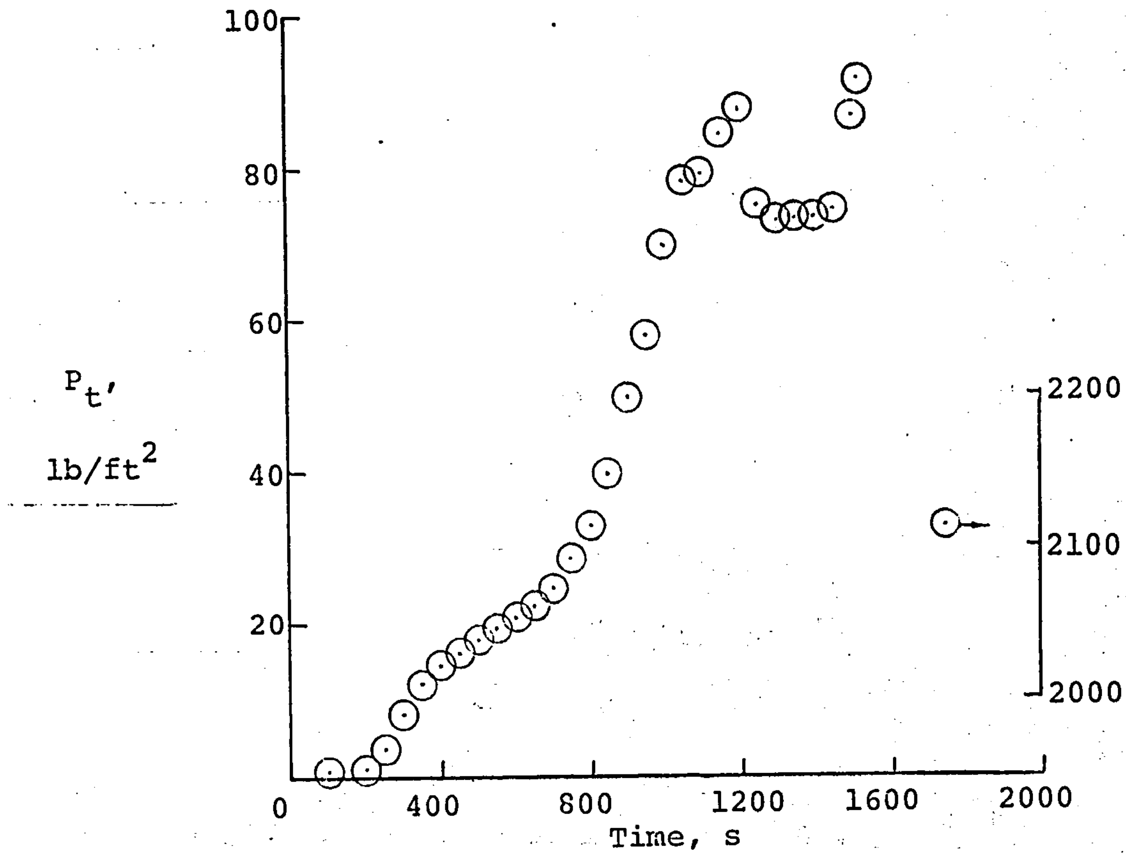


(e) Heating rate, body point 1702.

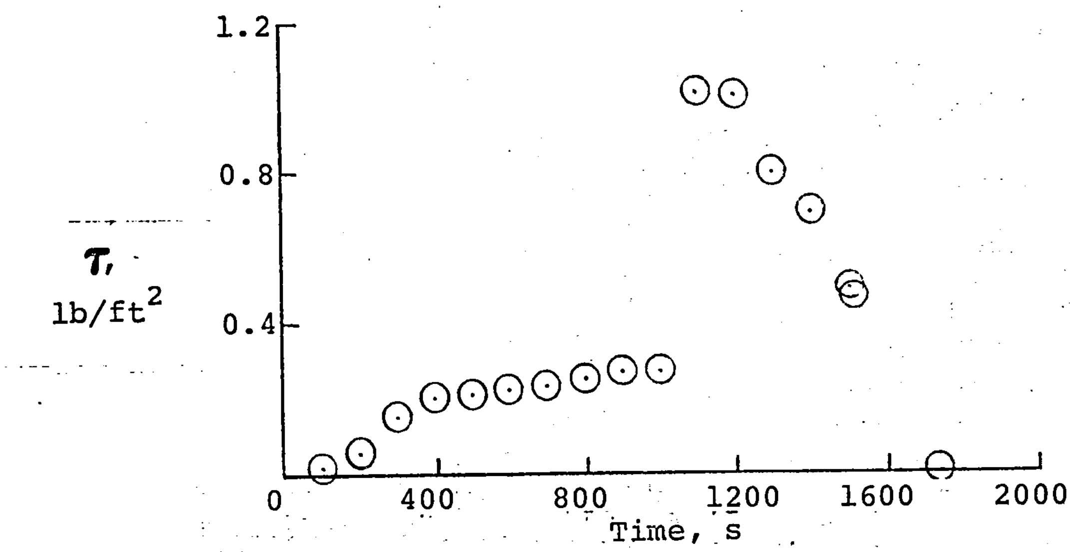


(f) Enthalpy, body point 1702.

Figure 7.- Continued

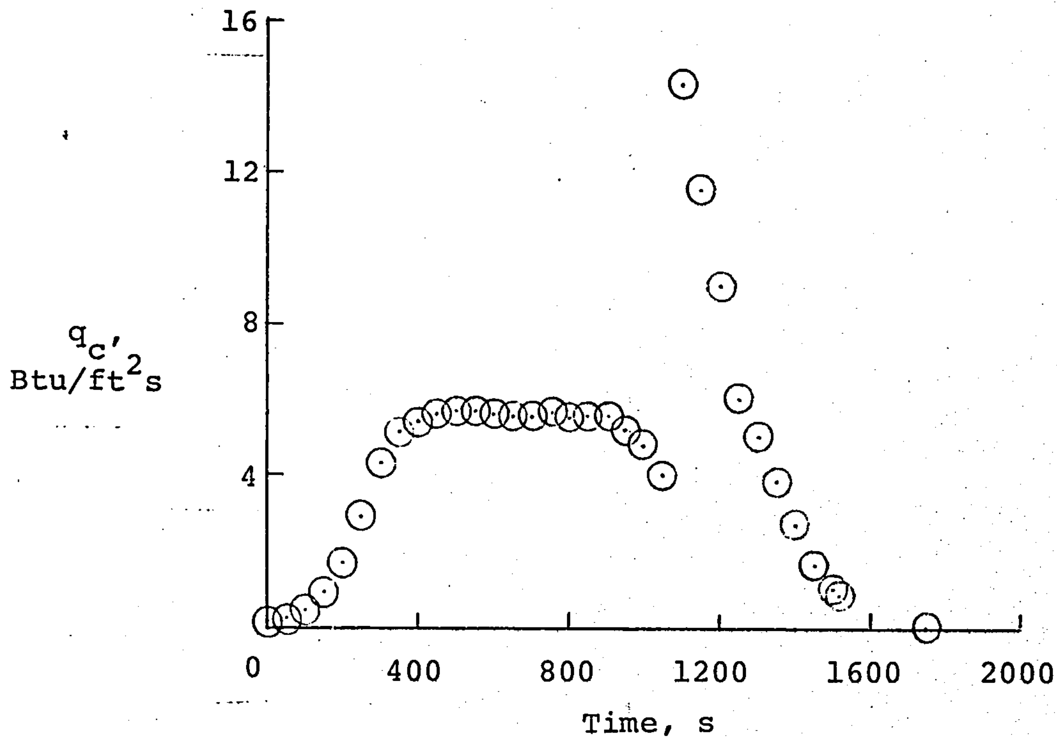


(g) Pressure, body point 1702.

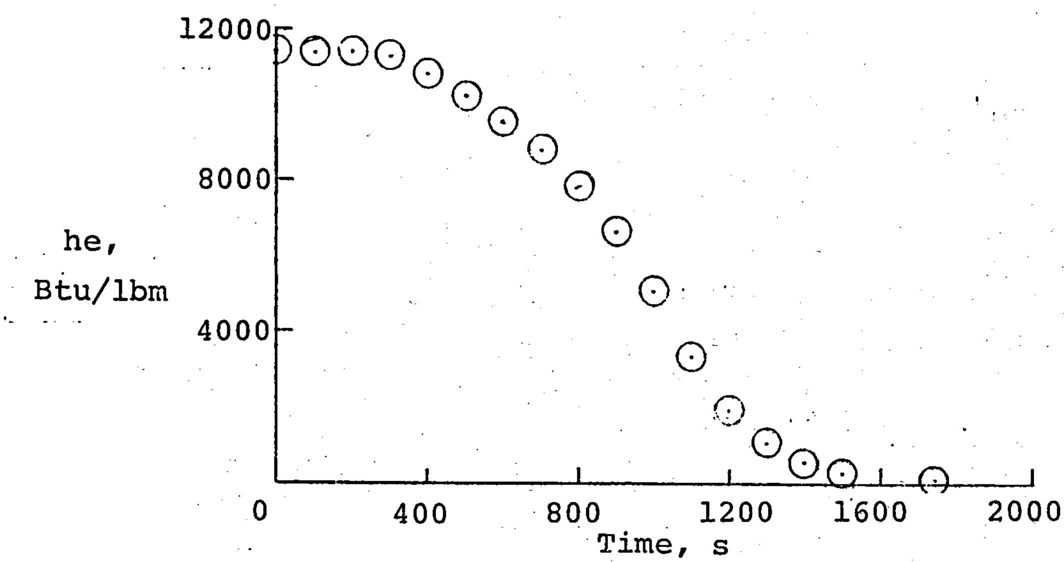


(h) Shear, body point 1702.

Figure 7.- Continued.

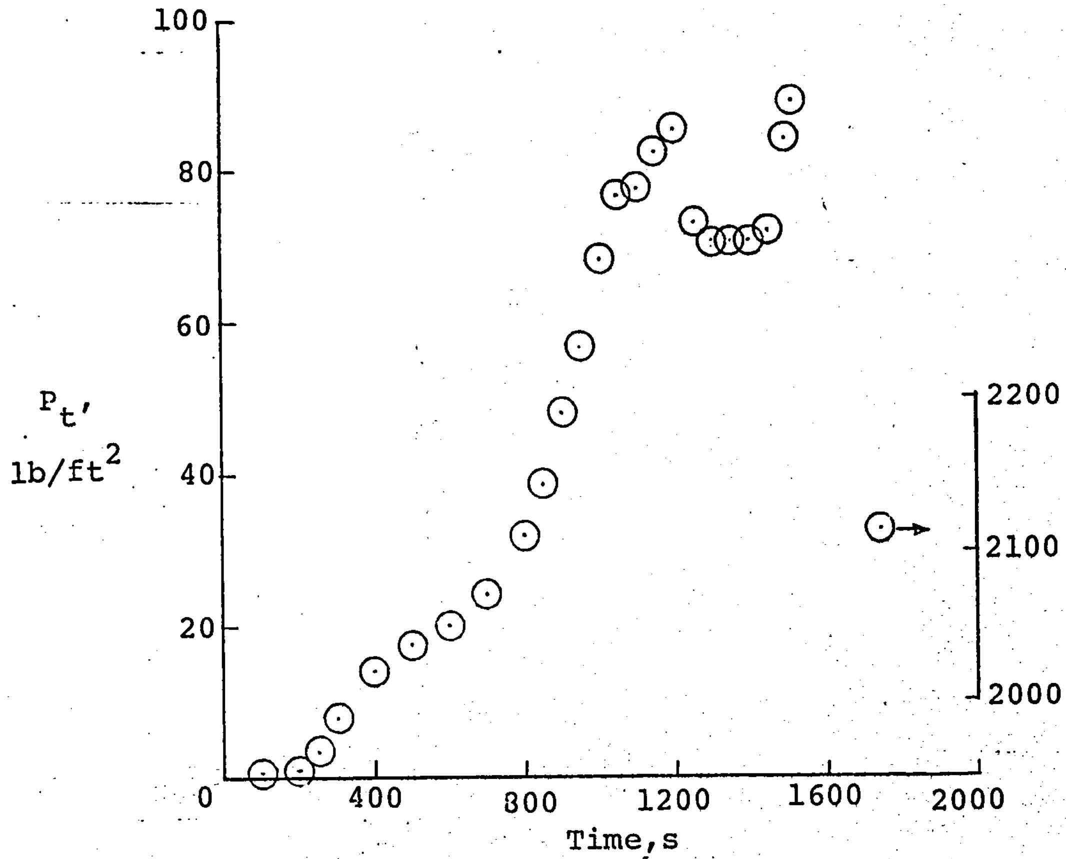


(i) Heating rate, body point 1800.

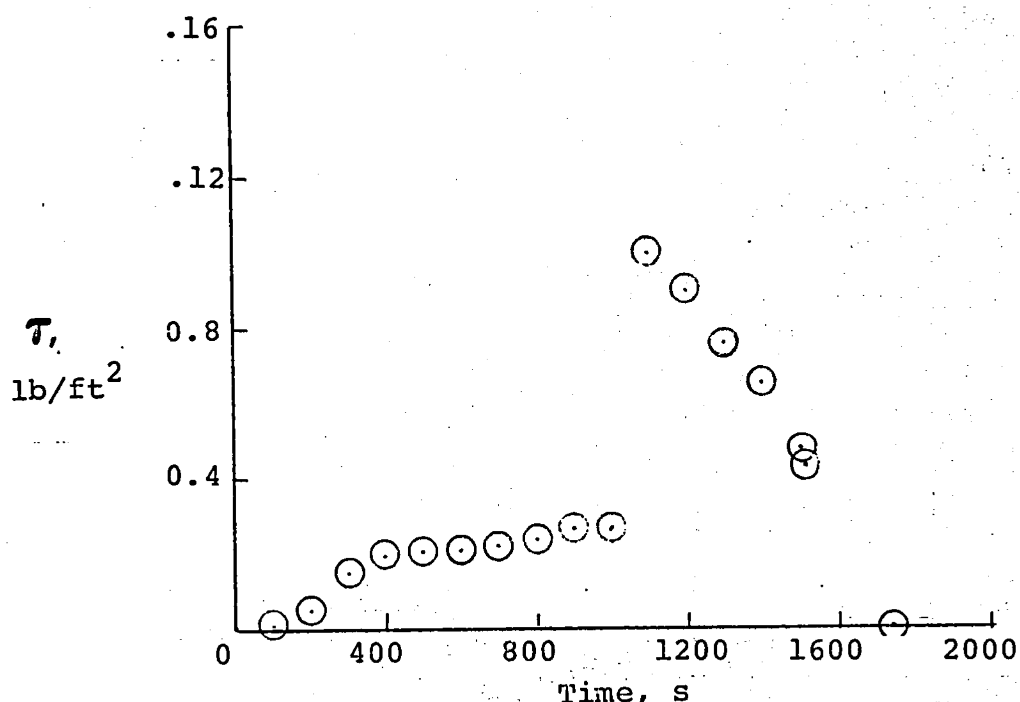


(j) Enthalpy, body point 1800.

Figure 7.- Continued.

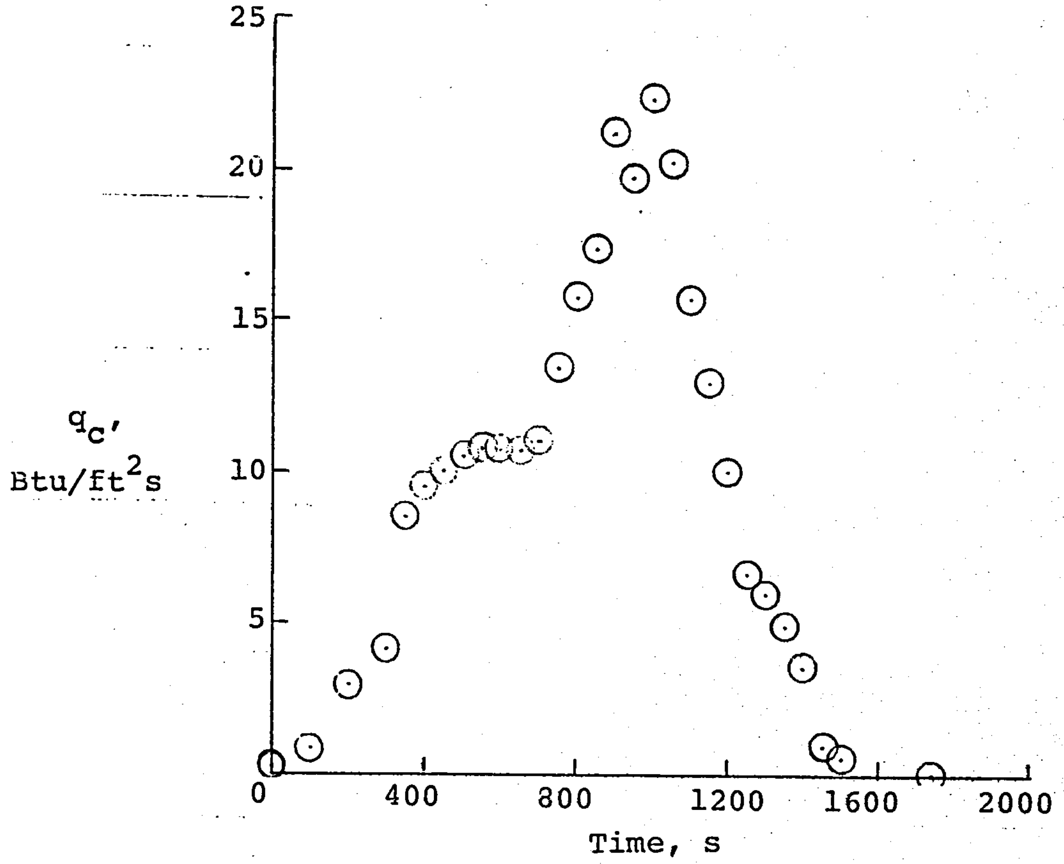


(k) Pressure, body point 1800.

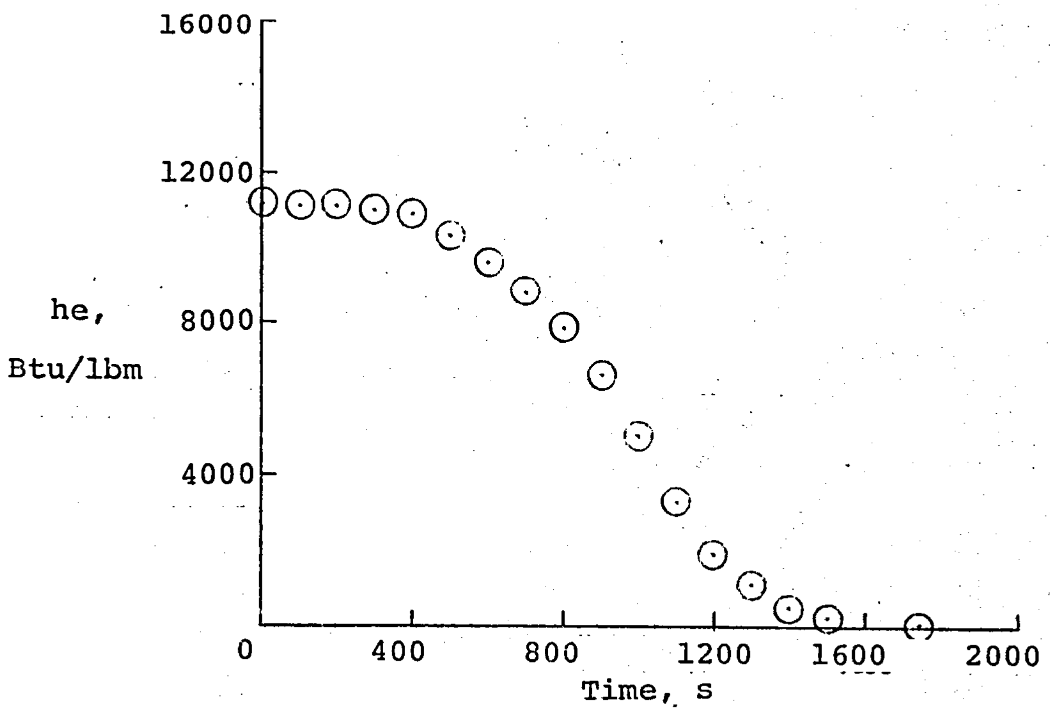


(l) Shear, body point 1800.

Figure 7.- Continued.

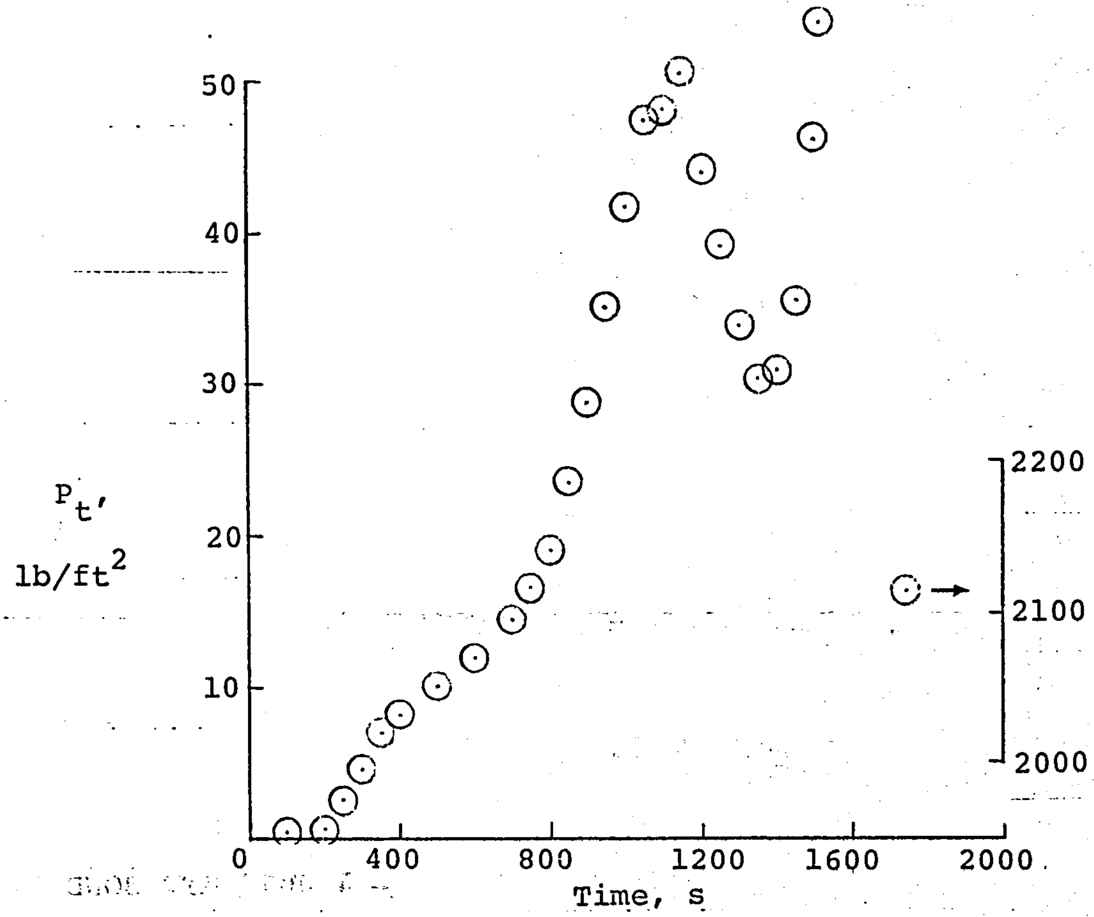


(m) Heating rate, body point 213.

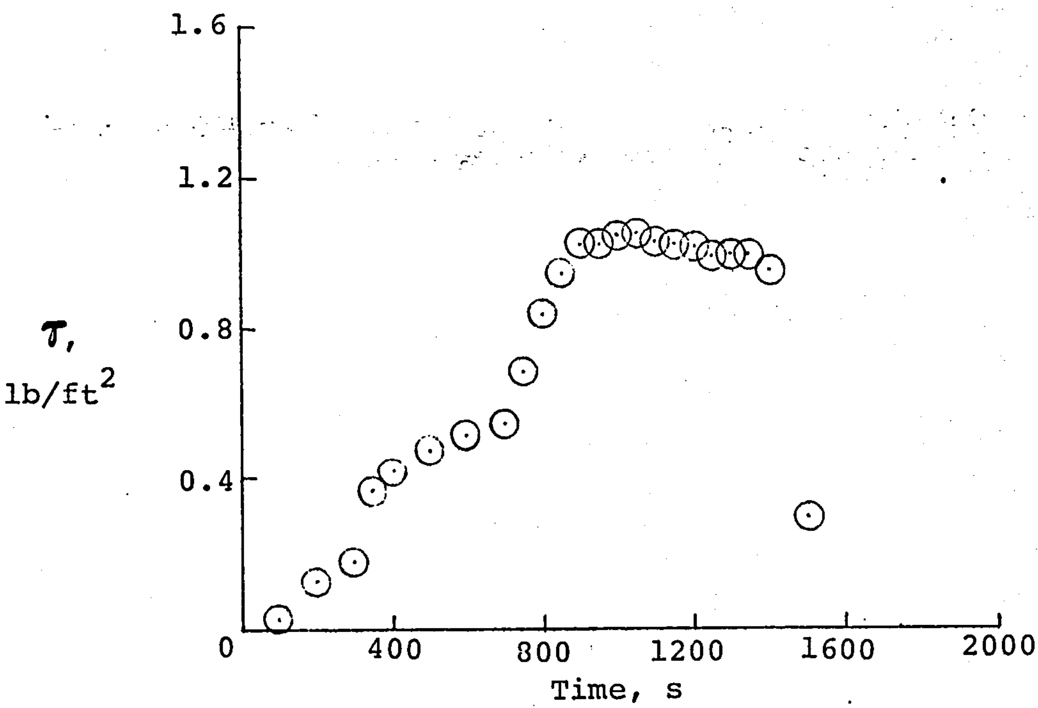


(n) Enthalpy, body point 213.

Figure 7.- Continued



(o) Pressure, body point 213.



(p) Shear, body point 213.

Figure 7.- Concluded.

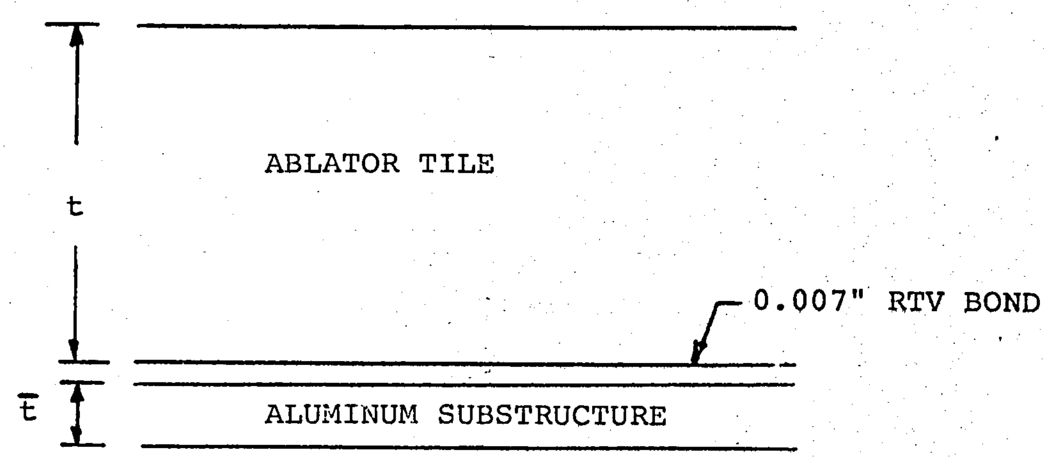
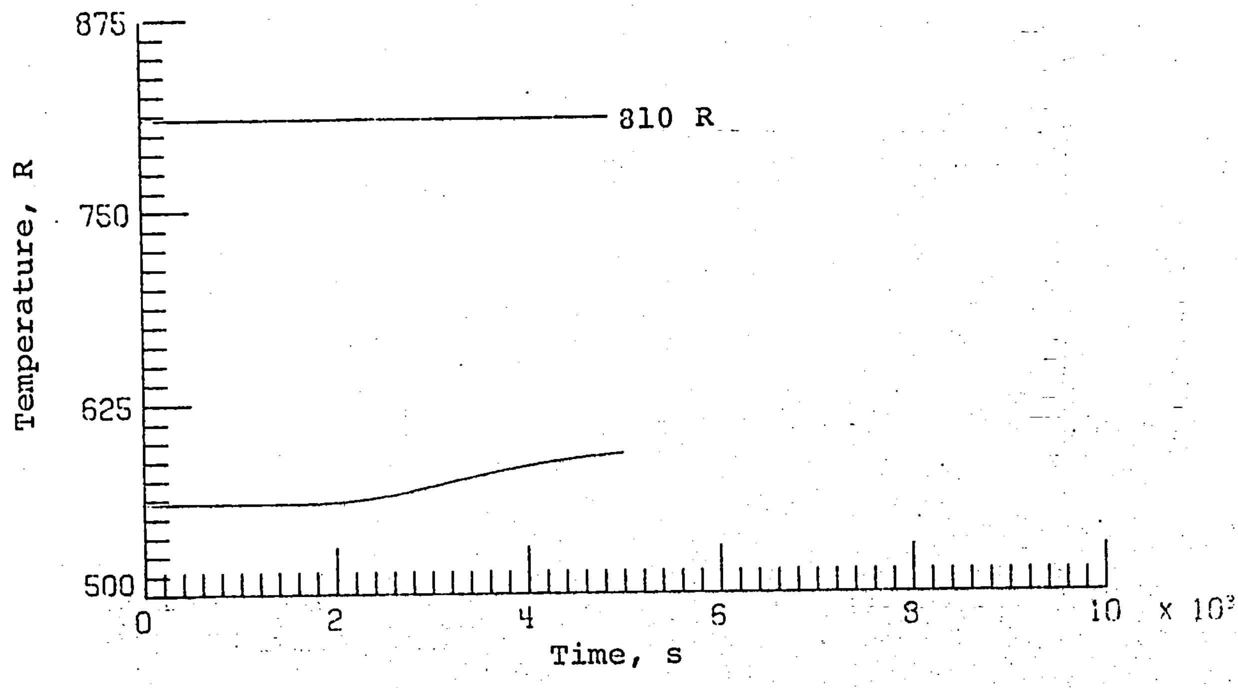
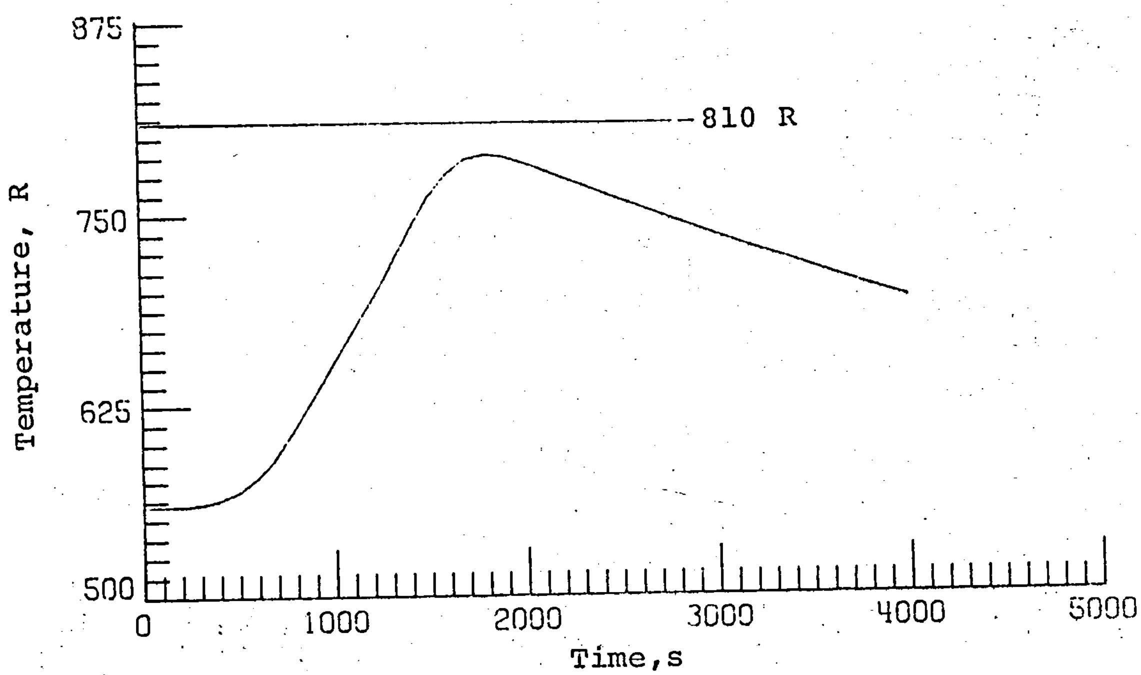


Figure 8.- One-dimensional model of ablator TPS used to calculate back surface temperatures.

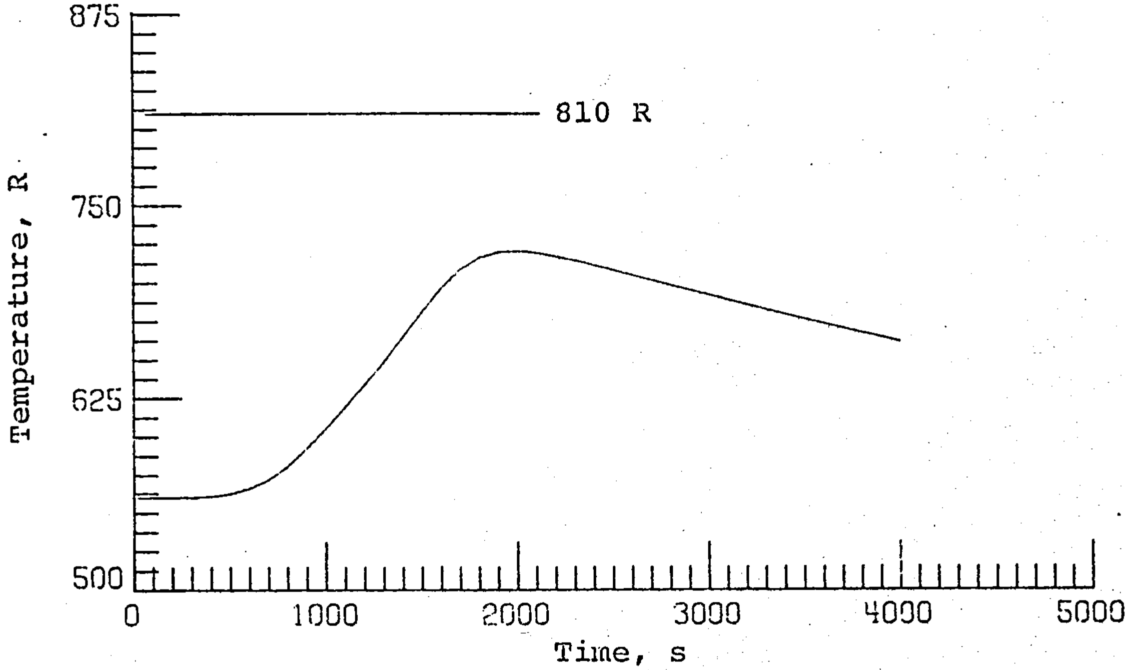


a) Body point 1030

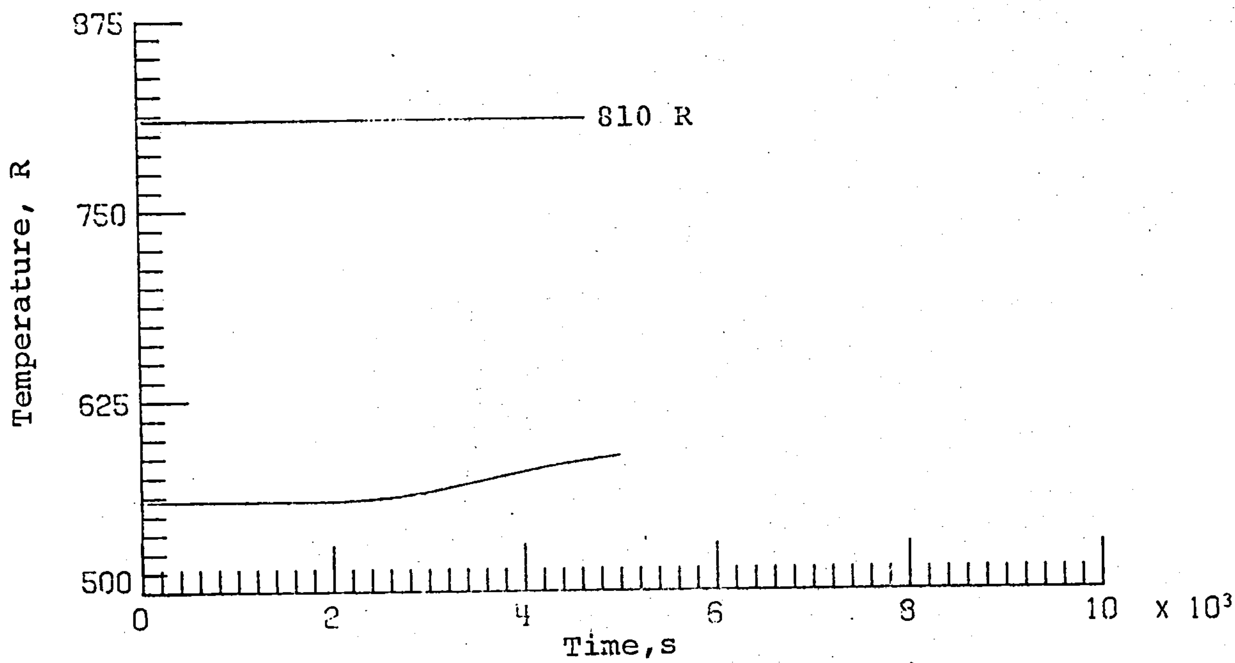


b) Body point 1702

Figure 9.- Back-surface temperature histories of an ablator (SLA 561) tile at several shuttle orbiter body points for the design entry trajectory 14414.1C.

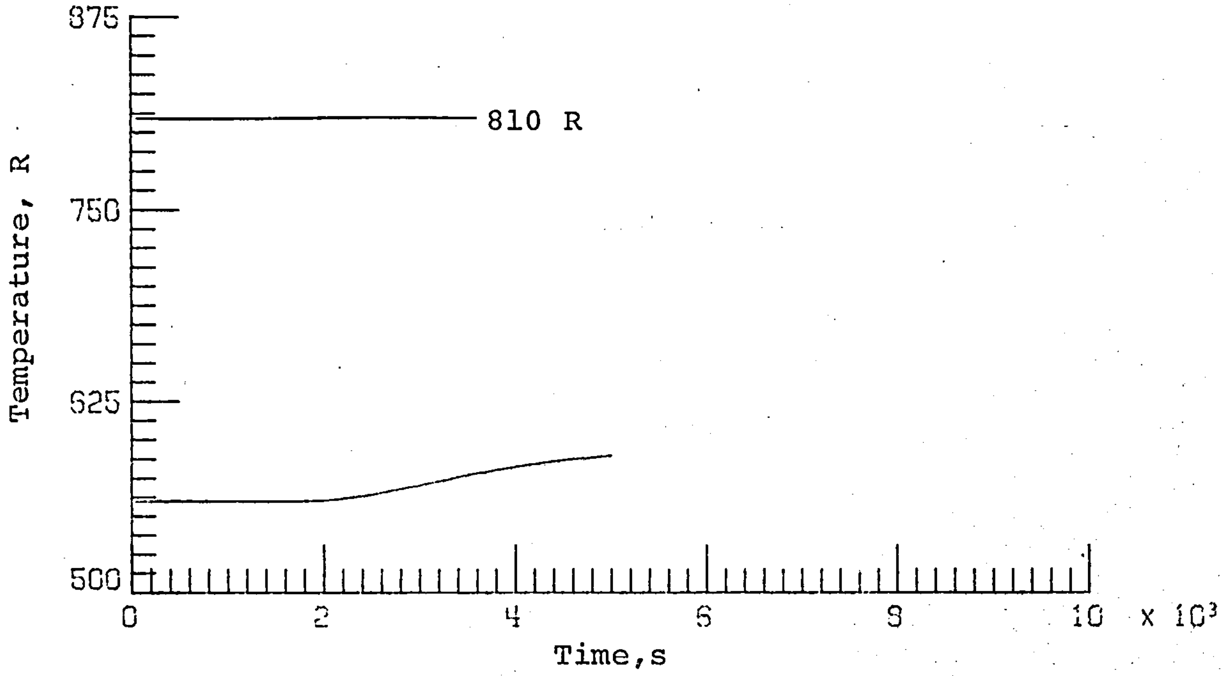


c) Body Point 1800

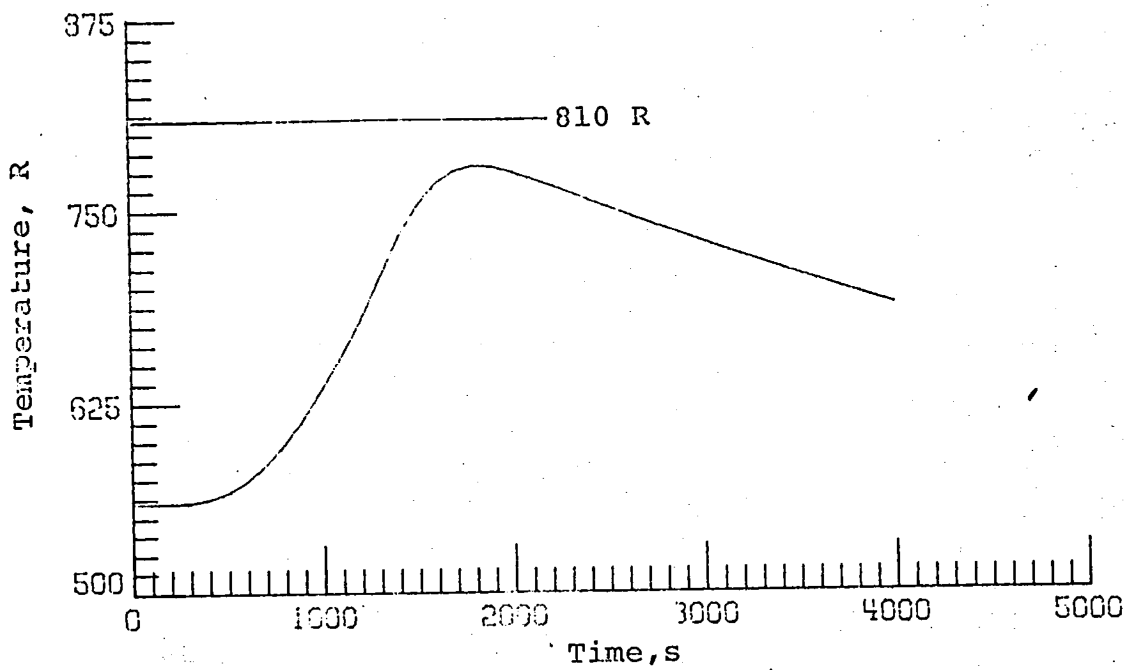


d) Body point 213

Figure 9.- Concluded.

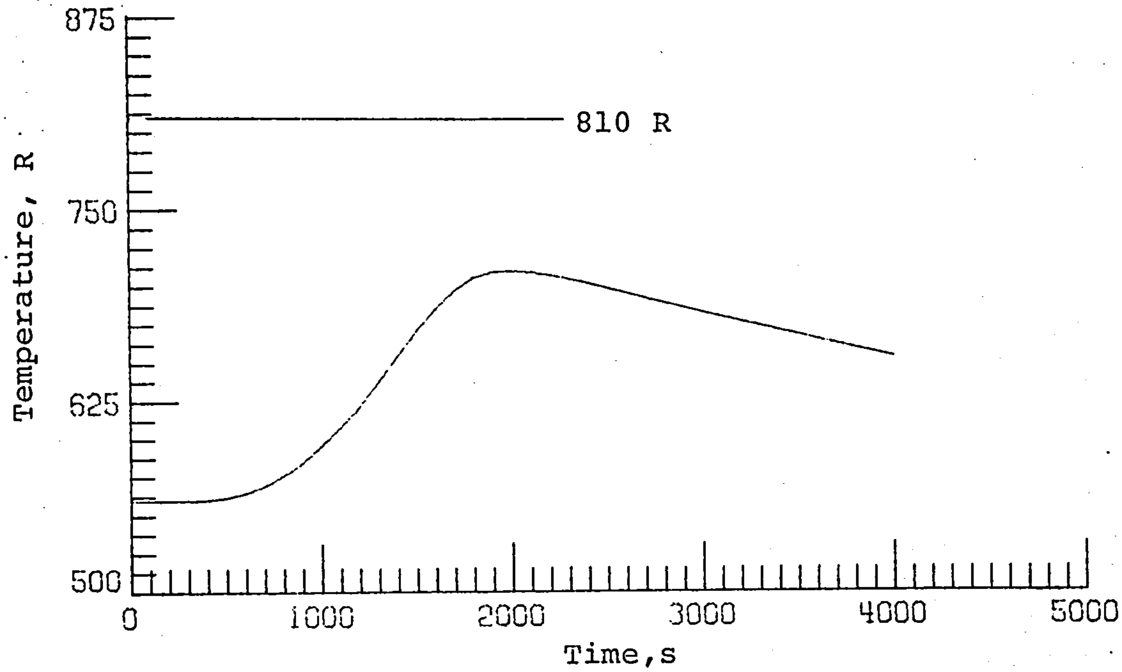


a) Body point 1030

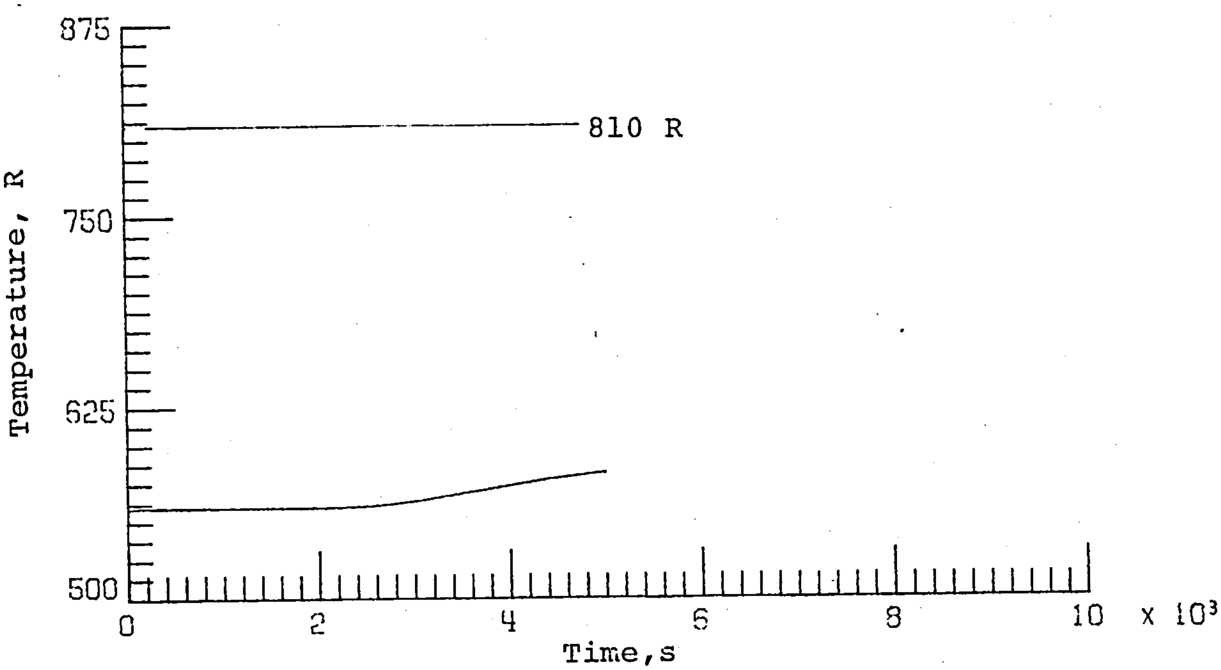


b) Body point 1702

Figure 10.- Back-surface temperature histories of an ablator (SLA 561) tile at several shuttle orbiter body points for the nominal entry trajectory STS-1.



c) Body point 1800



d) Body point 213

Figure 10.- Concluded.

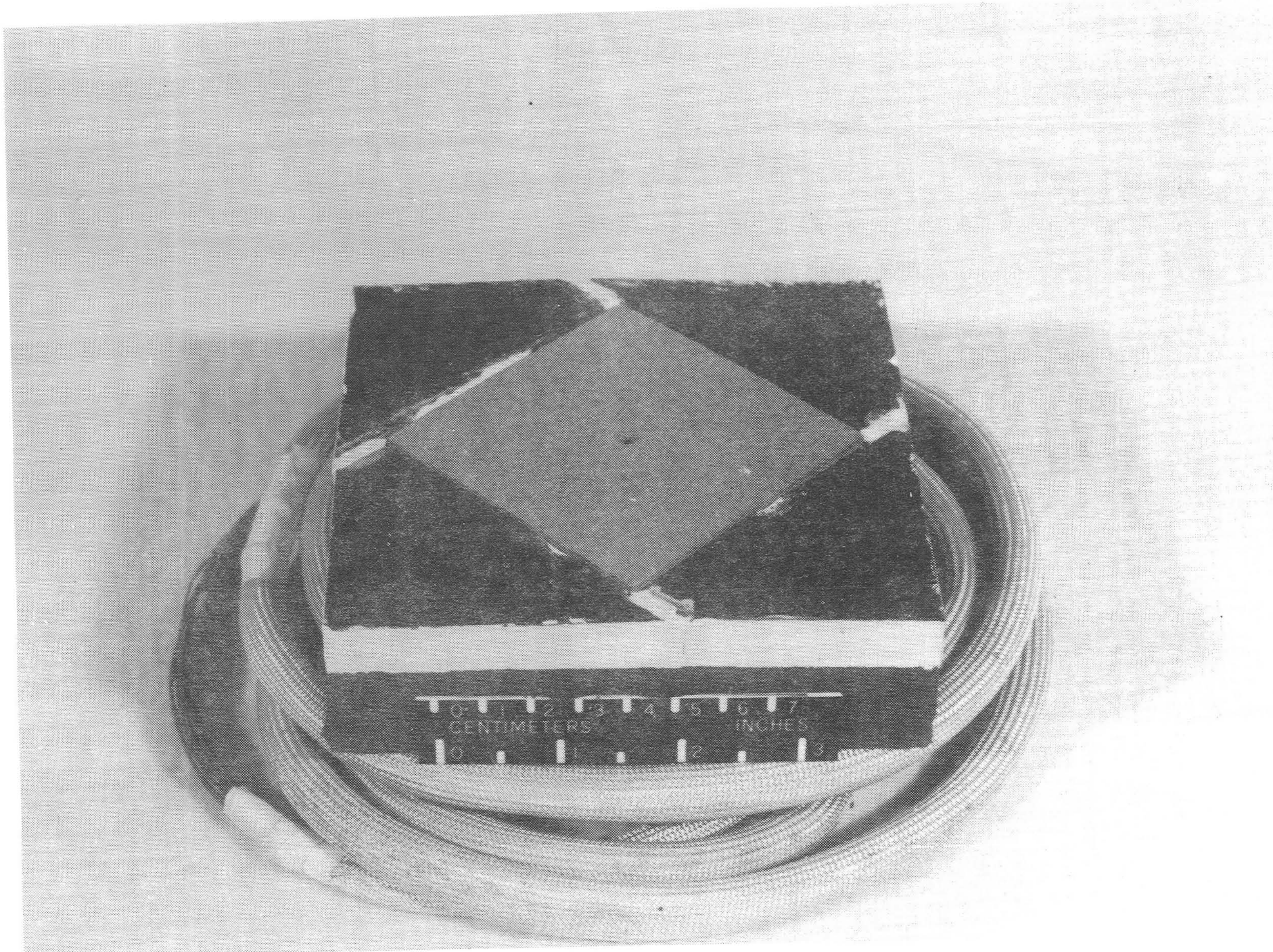


Figure 11.- Arc-tunnel test specimen consisting of SLA-561 ablator tile surrounded by RSI tile.

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16. Abstract
The shuttle orbiter relies primarily on a reusable surface insulation (RSI) thermal protection system (TPS). The RSI is very efficient in its thermal performance, however, the RSI tile system has shown poor mechanical integrity. In view of the apparently random features of the tile integrity problem, the dimension of the required effort to improve the integrity cannot be specified. Therefore, an investigation is needed of TPS systems which might be used to replace RSI on the shuttle. The ablative systems are far more highly developed than other alternatives, and are the only systems that can be considered for near term replacement of RSI.

The purpose of this paper is to review the state-of-the-art of ablative TPS by reviewing the work done as part of the shuttle technology program and to assess the readiness of ablators for use on the shuttle orbiter. Unresolved technical issues with regard to ablative TPS on shuttle are identified. Therefore, the NASA LaRC initiated short time, highly focused analytical and experimental programs to: (1) identify candidate ablation materials; (2) assess the data base for these materials; (3) evaluate the need and kind of waterproof coating; (4) calculate thermal and other stresses in an ablator tile; (5) identify an acceptable ablator/RSI tile joint filler; and (6) assess the sensitivity of the ablator to sequential heat pulses. Results from some of these programs are discussed.

17. Key Words (Suggested by Author(s))
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